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UH-2 HELICOPTER HIGH-SPEED FLIGHT RESEARCH WITH LIFT AND THRUST AUGMENTATION

W.E. Blackburn, et al

Kaman Aircraft Corporation
Bloomfield, Connecticut

October 1967

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USAAVLABS TECHNICAL REPORT 67-12

UH-2 HELICOPTER HIGH-SPEED FLIGHT RESEARCH WITH LIFT AND THRUST AUGMENTATION

By

W. E. Blackburn
A. D. Rita

October 1967

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA44-177-AMC-151(T)
KAMAN AIRCRAFT CORPORATION
BLOOMFIELD, CONNECTICUT

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This report has been reviewed by the U. S. Army Aviation Materiel Laboratories and is considered to be technically sound. The report summarizes the results of a flight research program to determine the high-speed flight characteristics of a multibladed, fully articulated, servo-flap controlled rotor system utilizing auxiliary jet thrust and wing lift augmentation. These results are published for the exchange of information and the stimulation of ideas.

The Army is currently continuing to sponsor other high-speed programs of a similar nature to provide basic technology for use in the design of future high-performance rotary-wing aircraft.

Task 1F121401A14311

Contract DA 44-177-AMC-151(T)

USAAVLABS Technical Report 67-12

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UH-2 HELICOPTER HIGH-SPEED FLIGHT RESEARCH WITH LIFT AND THRUST AUGMENTATION

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SUMMARY

This report presents the results of a flight test program conducted using a UH-2 helicopter with a fully articulated, servo-flap controlled main rotor and provided with a single YJ-85 jet engine for thrust augmentation and with wings for lift augmentation. The program consisted of an investigation of the effect of lift augmentation on airspeed limitations imposed by rotor blade stall or compressibility and an examination of the performance, flying qualities, structural loads, and vibrations within the envelope established.

A maximum level-flight speed of 190 knots was achieved. It was determined that the airspeed limit is imposed by total power available. Although compressibility effects were encountered at high rotor speed, the airspeed envelope was expanded by reducing rotor speed.

Lift augmentation is shown to substantially reduce rotor and airframe loads and vibrations while providing an expanded speed and maneuvering envelope. The suitability of a fixed-wing incidence angle was demonstrated, although it is concluded that further investigation of rotor and fixed-wing type controls may be required to provide optimum control both in level flight and in maneuvers.

Longitudinal dynamic stability has been improved somewhat by the addition of the wing in terms of short-term oscillations following a disturbance, although static stability with respect to speed has deteriorated slightly as compared with the jet-augmented UH-2. Dynamic lateral/directional stability has been improved compared to the jet-augmented aircraft because of the effective negative dihedral contributed by the wing installation. Inertial coupling due to configuration dissymmetry appears to have a significant influence on response to a simulated gust in the vertical or lateral direction, but the coupled accelerations are small and easily controlled.

Correlation of flight test results and those predicted by analytical study is presented in the areas of performance, trim and controllability, and limit flight speeds. In general, the analytical methods for predicting these characteristics are shown to be satisfactory.

FOREWORD

This report summarizes the results of a flight research program to determine the high-speed characteristics of a multiblade, fully articulated rotor system utilizing a UH-2 helicopter modified to provide lift and horizontal thrust augmentation. The program was conducted by the Kaman Aircraft Corporation, Bloomfield, Connecticut, under U.S. Army Aviation Materiel Laboratories Contract DA 44-177-AMC-151(T).

Research flights, which began in February 1965 and continued to completion of the program in April 1965, are a continuation of an investigation of methods of extending the high-speed capability of rotary-wing aircraft. Results obtained prior to this effort are presented in References 1 and 2.

This program was conducted under the technical cognizance of Messrs. L. H. Ludi and J. P. Whitman of the Applied Aeronautics Division of USAAVLABS. Principal Kaman Aircraft Corporation personnel associated with the program were Messrs. A. Ashley, G. Basile, W. Blackburn, E. Eckhart, H. McIntyre, W. Murray, A. Rita, F. Smith, and A. Whitfield.

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LIST OF SYMBOLS AND ABBREVIATIONS

SYMBOLS

Dimensional Quantities

A	main rotor disc area - feet^2
H_D	density altitude - feet
L_W	wing lift - pounds
L_R	rotor lift - pounds
N_R	rotor speed - percent
N_Z	normal load factor - gravity units
N_{ZR}	rotor load factor - gravity units
R/D	rate of descent - feet/minute
T	rotor thrust (compound) - pounds
T_J	net jet thrust of auxiliary propulsion engine (acting on aircraft) - pounds
T_S	rotor thrust (UH-2 without wings) - pounds
a_{1S}	rotor longitudinal flapping angle - degrees, positive for aft flapping with respect to the rotor hub
b_{1S}	rotor lateral flapping angle - degrees, positive for flapping to the right with respect to the rotor hub
g	gravitation acceleration unit - feet/second^2
i_W	wing incidence angle - degrees, positive trailing edge down with respect to aircraft waterline
Δf	added drag area of exposed wing - feet^2
α_F	fuselage angle of attack - degrees, positive nose up with respect to relative wind
δ_e	horizontal stabilizer deflection - degrees, positive trailing edge down with respect to aircraft water line

Non-dimensional Quantities

C_P	main rotor power coefficient
C_T	main rotor thrust coefficient
M_T	advancing tip mach number
c_d	drag coefficient
c_l	lift coefficient
μ	rotor tip speed ratio
σ'	air density ratio

ABBREVIATIONS

CAS	calibrated airspeed - knots
CG	aircraft center of gravity - fuselage station
GW	gross weight - pounds
GW_s	gross weight of a UH-2 helicopter without wings
HP	horsepower
IGE	in ground effect
MR	main rotor
OGE	out of ground effect
PSI	pounds/ $inch^2$
RPM	revolutions/minute
TAS	true airspeed - knots

CONFIGURATION DEFINED

Jet-Augmented

The use of these words throughout the report refers to the aircraft as shown in Figures 1 and 2 but without the wing and without the stabilizer trim control.

INTRODUCTION

A continuing research program to define the configuration of an aircraft which will combine good VTOL characteristics with high forward speed has resulted in the investigation of various schemes for extending the maximum speed attainable in a rotary-wing aircraft. Horizontal thrust augmentation and vertical lift augmentation have been shown analytically to be promising approaches to the problem.

From November 1963 until September 1964, a UH-2 helicopter with a YJ-85 jet engine installed was flown to investigate the effect of horizontal thrust augmentation on the limit airspeed envelope as defined by retreating blade stall or compressibility, flying qualities, control, stability, structural loads, and vibration at speeds well beyond the capability of the pure helicopter configuration. The results of this research program are reported in References 1 and 2. The present effort, the results of which are presented in this report, is a continuation of that research involving not only horizontal thrust augmentation but also lift augmentation provided by a fixed-incidence wing.

DESCRIPTION OF TEST VEHICLE

The aircraft used during this test, UH-2A BUNO 147978, was the same machine used for the investigation with thrust augmentation and is fully described in Reference 1. The wing installation, which consisted of the outer panels of the wing used on the Beech Aircraft Corporation Model 65 Queen Air, resulted in the configuration shown in Figure 1 and Figure 2. Also shown in Figure 2 is a tabulation of dimensional details pertinent to the UH-2 helicopter.

The wing panels were installed with no geometric dihedral angle, and the forward spars were located at fuselage station 176 below the cabin floor. The initial incidence angle of the root chord line with respect to the fuselage water line was 0 degrees, but provision was made for ground adjustment from 10 degrees trailing-edge up to 5 degrees trailing-edge down. The flaps were trimmable from the fully retracted position to a maximum deflection of 18 degrees trailing-edge down using the Beech motor, flexible drive, and actuators. Pilot control was achieved by means of a switch mounted on the collective stick. The aerodynamic surfaces, normally used as ailerons in the Queen Air, were employed as spoilers to reduce wing lift in autorotation. They were arranged to move in phase in the trailing-edge up direction and were controlled by a switch on the collective stick.

The stabilizer, which had a fixed incidence angle in the thrust augmented configuration, was modified for this investigation to allow an in-flight incidence angle change from 12 degrees trailing-edge up to 16 degrees trailing-edge down with respect to the aircraft water line. By this means the pilot was able to trim the aircraft to various angles of attack, thus obtaining a wide range of wing/rotor lift ratios at a given airspeed.

The wing carry-through structure necessitated removal of the aft fuel cells normally provided in the UH-2 helicopter. The fuel capacity thus lost was partially replaced by the internal tanks in the Beech wings. Total fuel capacity in the helicopter forward tank and the wing tanks was 196 gallons.

The trailing edge of the tail pylon was extended for additional directional stability. This was found to be desirable based on results obtained during the earlier investigation with thrust augmentation only.

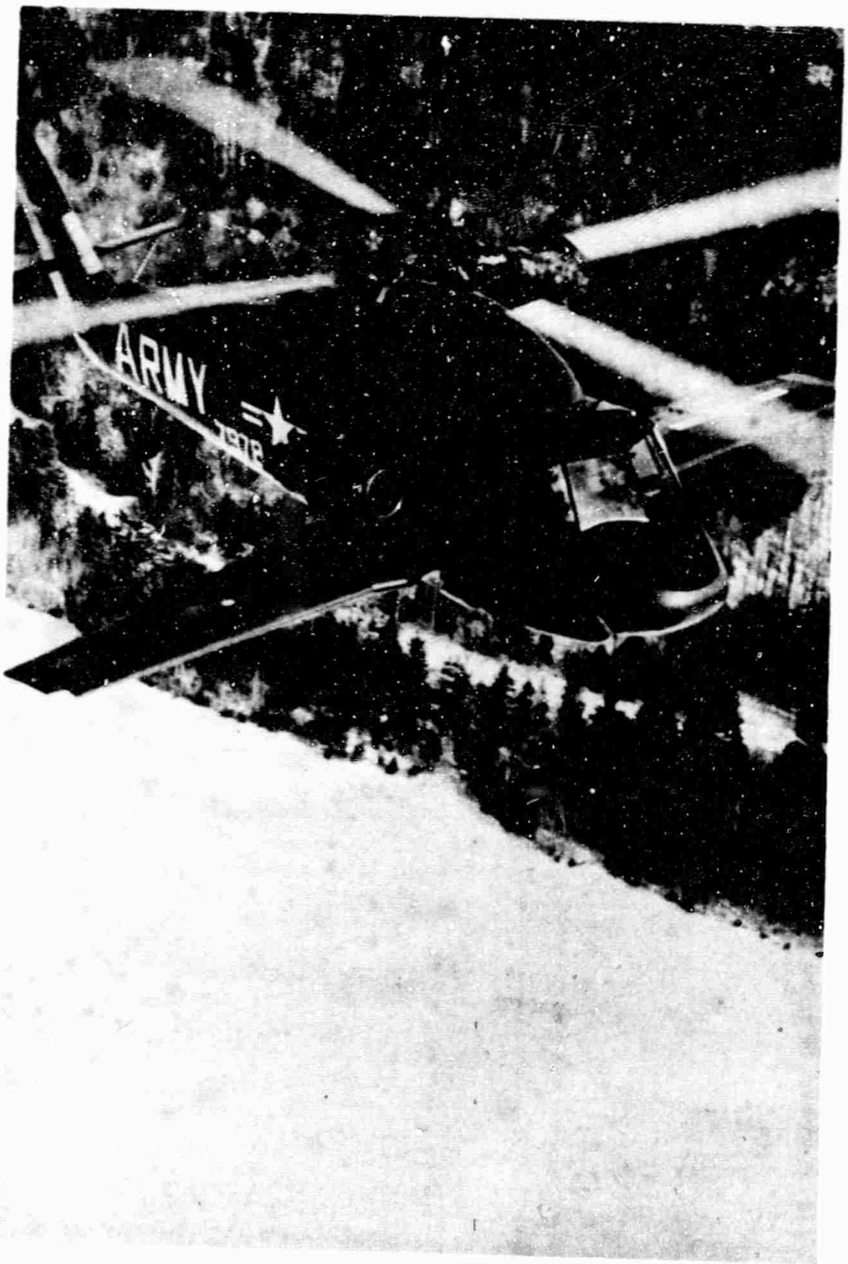

Figure 1. UH-2 Compound Research Helicopter.

Blade and Control Surface Areas	
Projected Disc Area	1520.50 sq. ft.
Total Blade Area Including Servo Flaps	134.00 sq. ft.
Servo Flaps, Total	9.98 sq. ft.
Horizontal Tail	14.50 sq. ft.
Vertical Tail	29.50 sq. ft.
Airfoil Sections	
Blade, Main Rotor	NACA23012 (Modified)
Blade, Tail Rotor	$NACA63_1$-012
Servo Flap Main Rotor Blade - $NACA63_3$-018 (27 deg. to -35 deg. Max Travel)	
Horizontal Tail - NACA0013 - Adjustable, Trailing Edge Dn. 16 deg., Trailing Edge Up 12 deg.	
Vertical Fin	NACA0025
Tail Rotor Surface Areas	
Projected Disc Area	50.4 sq. ft.
Total Blade Area	6.5 sq. ft.
Wing Area (Overall Exposed)	144. sq. ft.
Takeoff Gross Weight	10,200 lb.
100 Percent Rotor Speed is Equivalent to	276.7 RPM

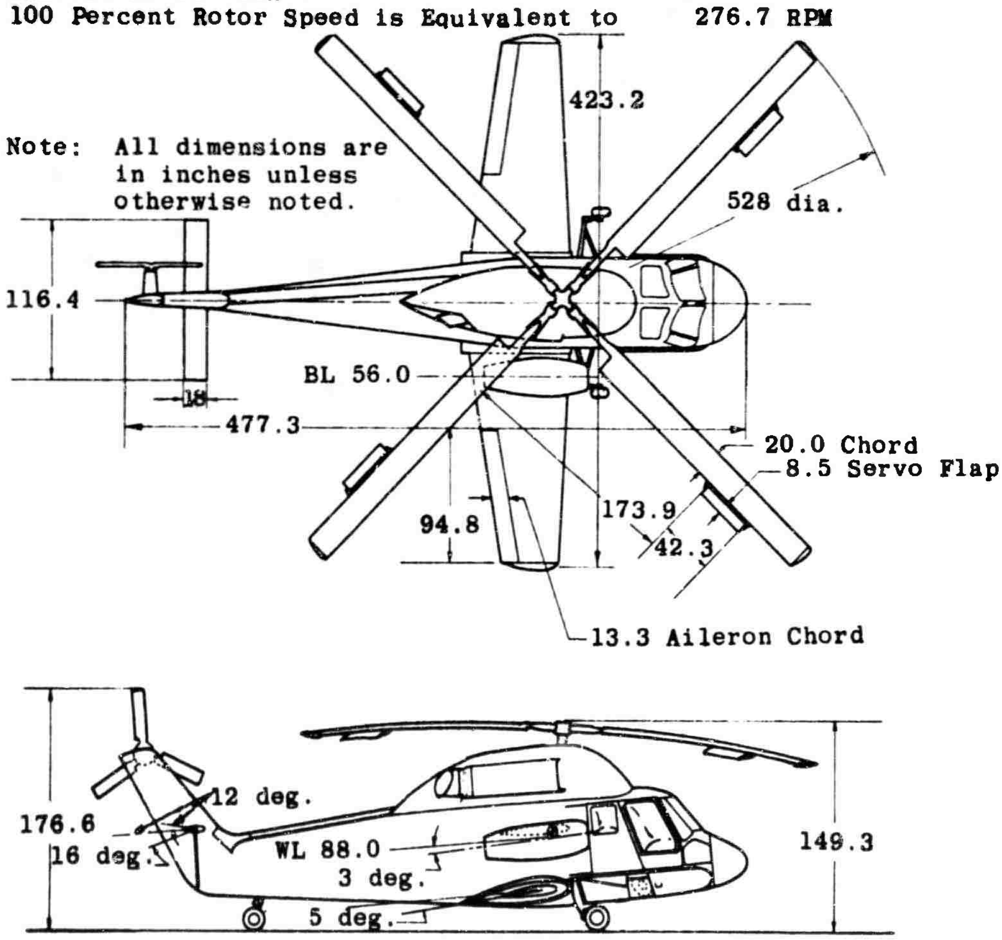

Figure 2. General Arrangement - UH-2 Compound.

TEST INSTRUMENTATION

Test instrumentation was installed to obtain flight data on the effect of wing and rotor load sharing on performance, controllability, stability, vibration, structural loading, and maneuverability.

Aircraft instrumentation consisted of a 9-channel telemetry system, two 36-channel recording oscillographs, and a 16-channel 35 millimeter photopanel together with the appropriate sensors for measuring the parameters listed in Table I. Dual instrumentation installations were used for control positions, main rotor flapping, and some vibratory loads to provide simultaneous telemetry monitoring and oscillograph recording during each flight. Continuous telemetry monitoring provided an instantaneous and comprehensive assessment of pertinent aircraft loads and vibrations throughout each flight to assure that the levels did not exceed the limits for safe operation.

An important requirement in the compound program was a suitable means of determining wing lift regardless of distribution. This requirement was satisfied using a system of strain gage bridges as described in Reference 3 at the root end of the wing spar which was sensitive only to wing shear load. Calibration during the wing proof-load test showed excellent agreement between the indicated and actual load.

Additional strain gages were installed on the wing, attachment fittings, and internal supporting structure to measure loads due to helicopter excited vibrations in the fixed-wing installation.

Accelerometers were used to measure the accelerations of the aircraft and YJ-85 engine. Additional velocity pickups were incorporated on the YJ-85 engine to determine its triaxial displacement and mode shape to supplement the acceleration data recorded in this area.

Wing tuft behavior was recorded by a motion picture camera during flights made to define the partial power descent and autorotative characteristics of the compound. General views of the instrumentation installation are shown in Figures 3 and 4.

TABLE I

SUMMARY OF PARAMETERS RECORDED

1. Longitudinal cyclic control position
2. Lateral cyclic control position
3. Stabilizer angle
4. Aileron angle
5. Directional control position
6. Collective control position
7. Wing flap angle
8. Tail rotor blade flapping
9. Tail rotor chordwise bending - Station 12
10. Tail rotor flapwise bending - Station 12
11. Main rotor hub flapping moment
12. Main rotor hub torque
13. Main rotor servo-flap bending
14. Main rotor chordwise bending - Station 43.5
15. Main rotor flapwise bending - Station 190
16. Pilot seat vertical acceleration
17. Tail pylon F&A acceleration
18. CG vertical acceleration
19. Transmission lateral acceleration
20. Station 50 vertical acceleration
21. Station 400 vertical acceleration
22. Monitor strain gages for wing bending
23. Wing attachment strain gages
24. Wing lift strain gages
25. Camera for wing tuft studies
26. Pitch attitude gyro
27. Roll attitude gyro
28. Sideslip angle
29. Yaw rate
30. Airspeed
31. Altitude
32. Rotor speed
33. Wing angle of attack
34. Critical transmission mount load - Tube 4
35. T-58 critical mount loads
36. YJ-85 critical mount loads
37. Horizontal tail flapwise bending
38. Horizontal tail chordwise bending
39. Horizontal tail aft spar load
40. YJ-85 compressor discharge pressure
41. YJ-85 thrust
42. T-58 gas-producer speed
43. YJ-85 gas-producer speed
44. CG vertical load factor
45. YJ-85 fuel flow
46. YJ-85 exhaust gas temperature
47. Outside air temperature
48. Main rotor control load
49. T-58 torque pressure

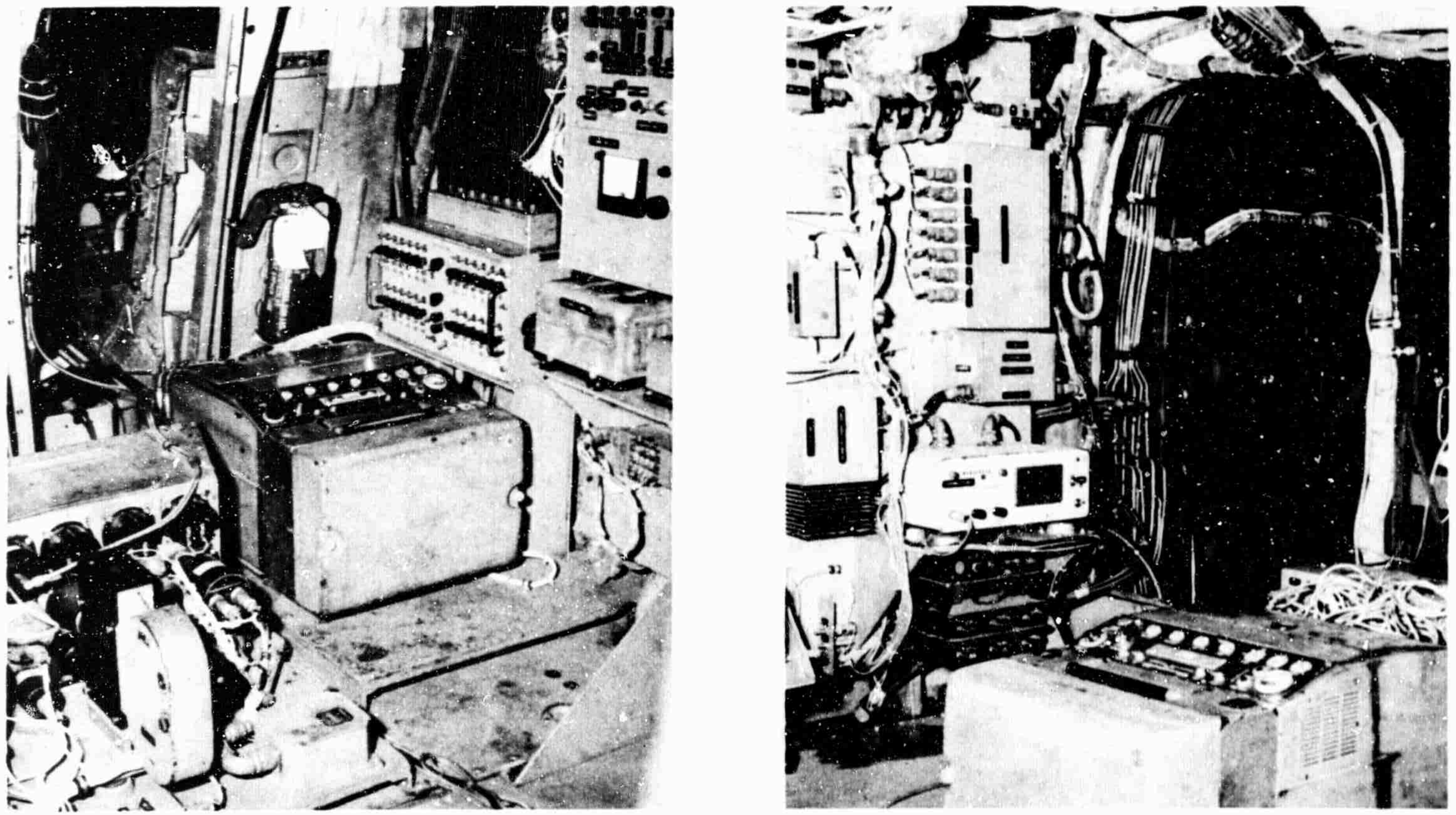

Figure 3. Overall Instrumentation Installation.

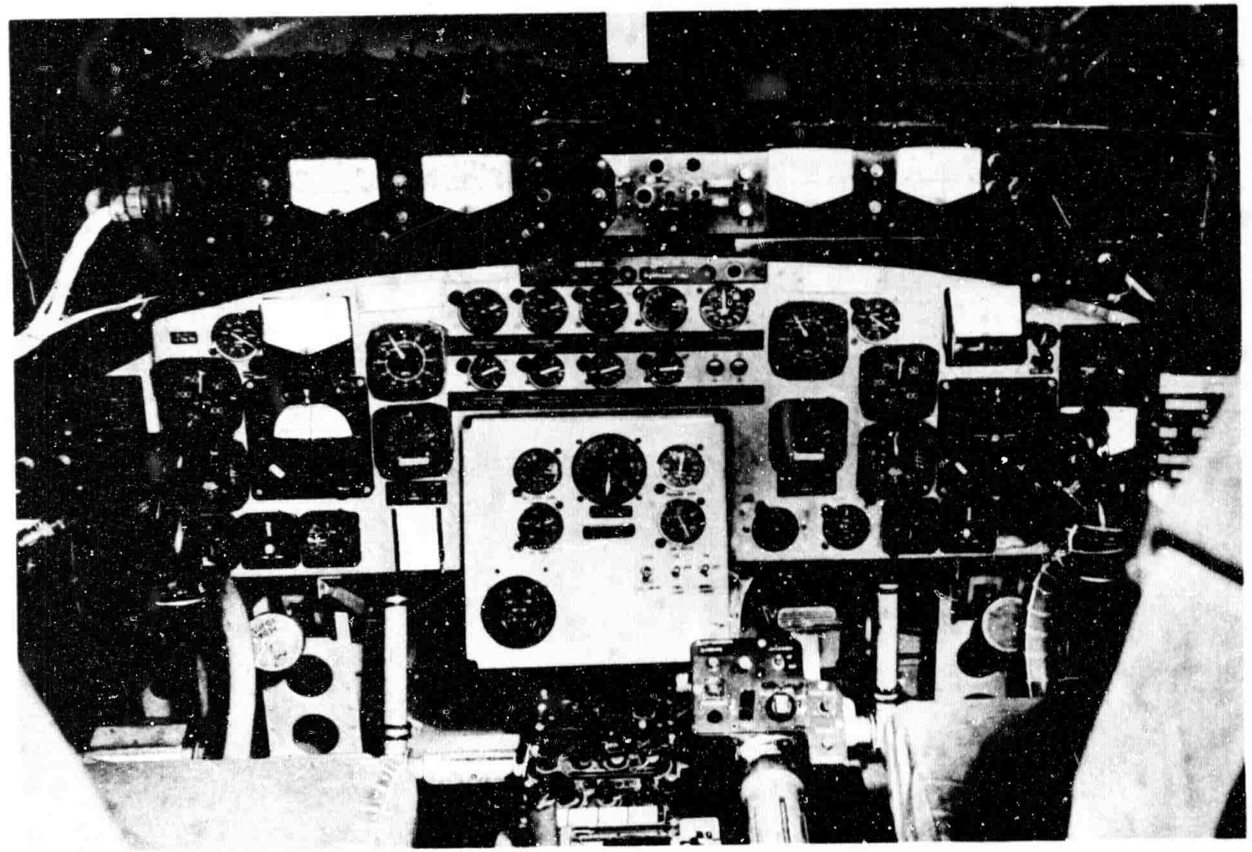

Figure 4. Compound UH-2 Helicopter Cockpit Instrument Panel and Visual Instrumentation.

EXPERIMENTAL PROCEDURES

Preliminary testing was conducted to substantiate the structural analysis of the compound design as presented in Reference 4. Structural proof-load testing was accomplished to 100 percent of the design limit load for the wing, horizontal stabilizer, and associated control systems. No excessive deflections or incipient failures were noted. Vibration testing indicated no resonant frequencies which would be excited in either the wing or the horizontal stabilizer installation.

Initial flight testing of the UH-2 compound commenced on 14 January 1964 and included an investigation of the ground handling and mechanical stability characteristics. No divergent tendencies resulting from the increased gross weight and roll inertia were noted. A wing incidence angle of 0 degrees was selected for the initial flight phase. Early flight testing investigated the general handling characteristics of the compound UH-2 at speeds of up to 100 knots with the jet engine secured. In addition, sideward flight, hovering turns, partial power descents, and climbs were investigated to obtain preliminary data on controllability, vibration, and structural loading. The vertical drag penalty imposed by the wing was examined for both IGE and OGE hover conditions.

A series of flights was then directed toward the development of an acceptable autorotation capability. The influence of spoiler deflection and stabilizer setting on autorotation was evaluated. Tuft studies and wing lift data were used to evaluate the behavior of the wing in autorotation.

Upon completion of the initial ground handling and autorotation investigations, a gradual level flight envelope expansion was conducted. During this phase of the program, data were obtained to evaluate performance, controllability, static stability, and vibration levels at true airspeeds up to 176 knots. The effect of varying wing lift was investigated by changing aircraft pitch attitude or deflecting the wing flaps. From the data obtained, it became apparent that at high airspeeds only a restricted range of wing lift could be examined without exceeding the main rotor flapping or longitudinal cyclic control limits. The deflection of wing flaps to obtain additional lift resulted in significant drag increases. Therefore, a change in the fixed-wing incidence angle to 5 degrees trailing-edge down was accomplished to obtain an increased range of wing lift without flaps, adequate control margins, and acceptable blade flapping relative to the rotor shaft. Subsequently, level-flight envelope expansion up to 190 knots TAS was achieved. Data were obtained at various airspeeds at two horizontal stabilizer incidence angles to determine the effect of wing lift on performance, rotor and airframe loads, vibration, and control requirements.

Main rotor stall and compressibility investigations were conducted from sea level to a 9000-foot density altitude for several values of jet thrust, wing lift, and rotor speed.

The static and dynamic longitudinal and lateral-directional stability characteristics were defined in level flight at airspeeds of up to 170 knots for several values of wing lift at 2400 pounds' jet thrust.

Maneuvers were performed to determine the load sharing characteristics of the wing and rotor as a function of load factor. Coordinated turns with bank angles of up to 60 degrees at airspeeds of from 140 knots to 170 knots with full jet thrust were executed during the program, as well as symmetrical pull-ups at 140 and 160 knots.

Flight testing was completed on 28 April 1965 after 70 flights involving 39.6 hours of aircraft time. A qualitative flight evaluation of the compound aircraft was made by USAAVLABS pilots at the Kaman Aircraft Corporation Flight Test Facility in Bloomfield, Connecticut, on 28 April 1965. A similar evaluation was made by flight test personnel from the Naval Air Test Center, Patuxent River, Maryland, on 21 May 1965.

EVALUATION

LEVEL FLIGHT PERFORMANCE

Figure 7 shows the total horsepower required to maintain level flight at various airspeeds as a function of wing lift. Test data are corrected to standard sea level conditions at a gross weight of 10,000 pounds for comparison with calculated results.

While the total power required reaches a minimum at different values of wing lift depending upon airspeed, it is essentially independent of wing lift in the range between 2500 and 4500 pounds. This behavior substantiates calculated trends at speeds of 160 knots and higher, but at lower speeds, calculations indicate that power is not affected by wing lift from 1000 to 6000 pounds.

The contribution of the wing to the overall drag of the aircraft consists of two parts, profile and induced drag, with the drag coefficient varying with the lift coefficient as illustrated in Figure 5. The minimum profile drag of the wing was responsible for an additional 2 square feet of equivalent flat plate area which is added to the 22.0 square feet obtained on the jet-augmented UH-2. The total drag contribution of the wing depends on the lift, and it is this factor which is most probably responsible for the small power variation observed in the test data. When the wing lift is small, the drag coefficient will be small as in region A of Figure 5. The rotor blade lift coefficient will be high, accompanied by a high drag coefficient as shown in region B. As the lift is shifted more and more to the wing at a given airspeed, the drag coefficient builds up, but at a slower rate than the blade drag coefficient drops off. The net effect is a decrease in overall power required. At high wing lift the opposite is true and a power increase may appear.

The actual value of the wing lift where minimum power is required is probably influenced also by changes in effective parasite drag area resulting from varied flow conditions around the wing/body intersection at the various fuselage angles of attack required to generate wing lift. A quantitative definition of this phenomenon would best be achieved by wind tunnel test, and it may be an important consideration affecting compound helicopter performance.

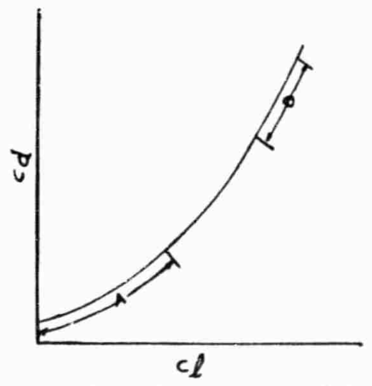

Figure 5. Variation of Drag Coefficient with Lift Coefficient (Schematic).

Theoretical calculations show that reasonable correlation with test data is obtained up to speed of 160 knots at 2500 pounds of wing lift. At higher speeds the analysis tends to be conservative in that it predicts a maximum speed of 177 knots, whereas the actual maximum speed obtained in flight test was 190 knots. Comparison of theoretical and measured results is shown in Figure 8.

In reviewing the analytical work, the conservatism in performance is attributed to the sensitivity of power required to a precise definition of the blade profile drag variation with angle of attack and the wing-body interference drag. Standard UH-2 performance has been calculated using techniques involving the assumption of a variable profile drag coefficient, and good correlation has been obtained within the speed range of the unaugmented UH-2. At higher speeds it was found that the assumed blade profile drag deviated substantially from the actual blade drag in the angle of attack range of interest. Using a more accurate representation of the profile drag, a reduction of 250 horsepower was calculated at 180 knots. On this basis, a maximum speed of 185 knots would be estimated for the compound helicopter.

A further improvement in the correlation of theory and test results would be achieved if the variation of flat plate area were taken into account. It is entirely possible that at the higher flight speeds the flow reattaches in the area of the wing root-fuselage intersection, thereby reducing the interference drag. The analytical method assumes a constant parasite drag area which can lead to error at high speed. At 180 knots, for example, a change of 1 square foot of flat plate area accounts for slightly more than 60 horsepower, which is equivalent to over 2 knots.

The deflected wing flaps were found to be responsible, as expected, for a substantial increase in power requirements in level flight. The comparison of power requirements with a clean wing and with flaps deflected 18 degrees is seen in Figure 8. The data indicate that the maximum speed obtainable with flaps deflected would be about 170 knots, compared to 190 knots actually achieved with a clean wing.

HOVER PERFORMANCE

Because of the increased flat plate area exposed to high-velocity downwash, a performance penalty in hover is to be expected. An estimation of the penalty can be made by using conventional hover theory. Calculation of nondimensional hover performance of the standard UH-2 from test data does not treat the flat plate area of the fuselage explicitly, but it is included in the thrust-to-power relationship where the thrust coefficient is based on the gross weight of the aircraft. Accordingly, the thrust required to lift the same weight of the compound configuration can be calculated from the following expression, which accounts for the additional vertical force resulting from impingement of fully developed downwash velocity on the exposed wing area:

$$T = \frac{T_S}{1 - \frac{\Delta f}{A}} \qquad (1)$$

The term Δf is the added drag area of the exposed wing and is equal to 144 feet^2; the rotor disc area (A) is 1520 feet^2 so that

$$T = \frac{T_S}{1 - 0.0948} = 1.10\ T_S \qquad (2)$$

The above expression indicates that the compound could be expected to require 10 percent greater rotor thrust because of the vertical drag of the wing. At equal thrusts, the expression $GW = GW_S(1-\Delta f/A)$ applies and indicates that $GW = .9052\ GW_S$ or a 9½ percent loss of lift due to the wing. Both of these expressions show good agreement with measured data for in and out of ground effect, shown in Figure 9, where the average gross weight penalty at comparative power levels is very close to that predicted.

AIRSPEED LIMITATIONS

The airspeed limitations encountered during this program were generally established by the advancing tip Mach number and by power requirements. However, with the wing at a 5 degree incidence angle with 0 flap deflection at maximum jet thrust available (2400 pounds), a power-limited true airspeed of 190 knots was achieved at 96 percent rotor speed corresponding to a tip speed ratio, μ , of 0.523. This test point, expressed as a calibrated airspeed of 189 knots at a density altitude of 250 feet, is shown on Figure 10 along with others obtained at higher altitudes which were limited by compressibility. Retreating blade stall was not experienced up to the maximum test density altitude of 9000 feet, attesting qualitatively to the effectiveness of the wing installation in relieving this limitation. Further evidence of this is seen by comparing the load factor achieved in maneuvers with the compound without encountering retreating blade stall with the stall-limited envelope previously established for the jet-augmented UH-2 (Figure 11). Although the seven maneuvers were performed primarily to examine wing/rotor load sharing characteristics as a function of normal load factor, it is noteworthy that, in three cases, the jet-augmented envelope was substantially exceeded with no indications of retreating blade stall.

In 14 cases the maximum airspeed attained appeared, from pilot observation, to be limited by the onset of compressibility on the advancing side of the disc. Analysis of these particular test points showed that in nearly all cases the actual tip Mach number exceeded that which would be predicted at a given main rotor power level when corrected for rotor thrust and tip speed ratio according to the relationships shown in Figure 50 of Reference 1. The comparison of the actual advancing tip Mach number with the predicted value is presented in Table II. These results probably reflect the

pilot's willingness to penetrate somewhat more deeply beyond the onset of compressibility because of the improved handling characteristics of the aircraft brought about by the installation of the wing.

AIRCRAFT TRIM AND CONTROL IN LEVEL FLIGHT

The variation of wing lift is shown in Figure 12 as a function of airspeed and horizontal stabilizer incidence angle. The individual points show the measured wing load, and the continuous curve represents calculated lift at the angles of attack computed from the aircraft attitudes shown in Figure 13, using appropriate airfoil characteristics, wing area, aspect ratio, and washout. The results shown in Figure 12 are corrected to include a fuselage carry-over factor of 1.11 estimated for the UH-2 wing/fuselage combination which accounts for the lift contributed by the wing area blanketed by the fuselage. Thus, the curve may be used to define wing lift for any straight and level trimmed flight condition where the stabilizer angle and airspeed are known.

The flight program was undertaken initially with the wing incidence angle set at 0 degrees relative to the fuselage water line. It became evident that, in order to develop significant wing lift at high speeds, the attitude of the helicopter, which determines the wing angle of attack, would necessarily be nose-up relative to the flight path requiring forward rotor flapping approaching the endurance limit as indicated in Figure 14. It was concluded, therefore, that increasing the wing incidence angle to 5 degrees trailing-edge down relative to the fuselage water line would permit investigation of a wider range of wing lift at high airspeeds. Subsequent testing at the higher incidence angle substantiated the predicted decrease in forward flapping, with the fuselage attitude remaining well within acceptable limits at significant levels of rotor unloading, as shown in Figure 13. Based on these test results and the effect of thrust augmentation on rotor flapping reported in Reference 1, it can be concluded that with higher levels of thrust augmentation relieving the requirement for the rotor to supply propulsive force, a substantial extension of the airspeed spectrum may be achieved with this aircraft.

The longitudinal cyclic stick position variation with airspeed is illustrated in Figure 15 for 2 stabilizer incidence angles. Note that the gradients shown were obtained with fixed stabilizer angles and varying wing lift. Wing lift varies, as previously shown in Figure 12. The maintaining of constant wing lift requires a movable stabilizer which will result in a different longitudinal cyclic control gradient. For example, Figure 12 shows 4100 pounds of wing lift at 150 knots with a stabilizer incidence angle of 6.8 degrees. Figure 15 shows that the corresponding longitudinal cyclic stick position is 50 percent of full travel. Increasing airspeed while maintaining wing lift constant requires a decreasing stabilizer setting (2.2 degrees at 170 knots, Figure 12), and the corresponding longitudinal cyclic stick position is 75 percent

of full travel (Figure 15) compared to only 70 percent if the stabilizer incidence angle had remained fixed at 6.8 degrees. Constant wing lift represents a 25 percent increase in the slope of longitudinal cyclic stick position over that associated with the same variation in airspeed and a fixed stabilizer setting.

At about 55 percent collective, longitudinal cyclic input is induced by further application of collective, which is inherent in the basic control system and results in the flattened stick position curve seen in Figure 15. Without the effect of this collective-induced longitudinal cyclic, Figure 15 also indicates that airspeed would be control-limited at about 190 knots. Higher airspeed attained with additional jet thrust would eliminate the cyclic control advantage from collective induced input, but this would be nearly compensated for by the reduced longitudinal cyclic requirement resulting from the added thrust as typified in Figure I-30 of Reference 1.

Collective stick positions for trimmed level flight, presented in Figure 16, show adequate margins within the power capability of this aircraft. The effect of increased wing lift is to decrease the collective requirement, since wing lift change is analogous to a gross weight change as it affects the rotor.

The substantial increase in collective requirements for the compound configuration noted in the airspeed range between 150 and 190 knots is attributed to the lower collective rigging of the compound and the decreasing wing lift as airspeed builds up. The collective stick position for a given blade pitch angle was raised for the compound, as will be explained in the section covering autorotation characteristics (page number 21). At speeds above 190 knots, ample collective control will be available, since both the thrust and the wing lift required to obtain higher speeds will significantly decrease collective requirements.

Lateral cyclic stick requirements are shown in Figure 17. The effect of wing lift does not appear to be significant. Comparison of the lateral cyclic trim positions of the compound and the jet-augmented helicopter shows a 10- to 15-percent additional increment of right cyclic for the compound. This is attributed to the collective/lateral mechanical mixer which is an integral part of the UH-2 control system. Its function is to relieve the increasing left cyclic stick input required as forward speed increases by inducing left lateral cyclic with increasing collective stick position. The lower collective rigging of the compound helicopter noted earlier requires a higher collective stick position for a given rotor thrust than the standard UH-2 or the jet-augmented model. This higher collective stick position mechanically introduces greater than normal amounts of left lateral cyclic for which the pilot must compensate with right stick.

The most critical control requirement was directional control pedal at speeds of 140 knots or below with maximum horizontal thrust augmentation. Directional control pedal displacement as a function of airspeed is plotted in Figure 18, where it can be seen that the right pedal requirement is maximum at speeds below 140 knots with full jet thrust augmentation. At speeds above 140 knots, more main rotor power must be supplied which relieves the right pedal requirement.

The analytical program defining trim attitude and control utilized a parabolic variation of rotor profile drag coefficient with lift coefficient which was subsequently found to be high compared to the appropriate NASA airfoil section data. Consequently, correlation of test and analysis on an absolute basis would not be expected. However, changes in trim parameters should show good correlation at constant airspeed where the effect of profile drag is minimized. On this basis, Figures 19 and 20 are presented to illustrate the effect of horizontal stabilizer deflection on fuselage attitude and longitudinal flapping at constant jet thrust at various airspeeds. Figures 21 and 22 show the effect of both stabilizer deflection and wing lift on longitudinal cyclic control. The extent to which the agreement between test and analysis is obtained confirms the adequacy of present analytical methods to define the steady-state flight requirements of a compound helicopter and further suggests that a more precise definition of rotor drag characteristics will result in satisfactory definition of all trim parameters on an absolute basis.

STATIC LONGITUDINAL STABILITY WITH RESPECT TO SPEED

The static longitudinal stability with respect to speed is somewhat more negative for the compound than for the jet-augmented helicopter. This would be expected because of the dependence of the rotor static stability on the magnitude of the lift vector. This characteristic is noted on Figure 23, which shows the stability exhibited by the compound at high and low elevator settings at a trim airspeed of 160 knots. At higher wing lift (2.2-degree trailing-edge down stabilizer incidence angle), the aircraft is slightly more unstable than at a lower wing lift. At 140-knot trim airspeed, the difference in stability is not evident because of the small difference in wing lift at the two stabilizer settings.

STATIC LATERAL/DIRECTIONAL STABILITY WITH SIDESLIP ANGLE

Directional control pedal displacement as a function of sideslip angle, shown in Figure 24, is essentially unaffected by either airspeed or wing lift. Elimination of the small angle band of neutral stability observed in the jet-augmented configuration is attributed to the increased chord of the tail rotor pylon of the compound.

Static lateral/directional characteristics are strongly affected by the wing installation. The relationship between directional control pedal and lateral cyclic stick displacement from trim,

plotted for various sideslip angles in Figures 24 and 25, shows negative stability in that right stick is required with a right pedal input. This represents a change from the characteristics measured on the jet-augmented helicopter, which exhibited positive lateral/directional stability, and results from a change in the effective dihedral angle from positive to negative. Although the wing was installed with 0 geometric dihedral angle, it is apparent that effective anhedral resulted due to the influence of flow around the fuselage on the low-wing installation.

No handling difficulty in normal maneuvers or coordinated turns resulted from the negative static stability characteristics of the test vehicle.

LATERAL/DIRECTIONAL DYNAMIC STABILITY

Side gusts were simulated by pulse inputs of directional control. Results are presented in Figures 26 through 28 for the conditions investigated. Lateral cyclic inputs required to control the magnitude of the roll angle resulting from sideslip in the jet-augmented configuration are substantially reduced for the compound, due to the decreased dihedral and the increased roll inertia afforded by the wing installation. A further significant effect of this change was the elimination of the Dutch roll motion noted at high speed and high gross weight in the configuration without wings.

The effect of airspeed on the characteristic motion about the vertical axis following a disturbance is seen by comparing the 140- and 170-knot conditions presented in Figure 26 at 2.2-degree trailing-edge down stabilizer incidence angle with 2400 pounds of thrust augmentation. At either speed, the motion is essentially dead-beat in yaw. The effect of wing lift on the response to simulated side gusts is small. Data are presented in Figure 27 showing left pedal inputs at 160 knots with 3300 and 4750 pounds of wing lift and 2400 pounds of thrust augmentation. The yaw response appears to be essentially the same at either value of wing lift.

The control-fixed response of the aircraft to side gusts demonstrates a pitch-yaw coupling which is best seen by comparing the 160-knot cases shown in Figure 28 for left and right pedal inputs with 2400 pounds of thrust augmentation at a constant wing lift. The left pedal input produced right sideslip followed by nose-up motion of the aircraft. With right pedal input, no pitching motion resulted until the aircraft swung back to the left, when a nose-up pitching motion of sufficient magnitude to require recovery control was experienced. The coupled accelerations are small, requiring only moderate corrective control to eliminate them, as shown in Reference 2 where an effort was made by the pilot to obtain pure motion about a given axis.

LONGITUDINAL DYNAMIC STABILITY

The response of the helicopter to a vertical gust input was examined by simulating the gust with a pulse input of longitudinal cyclic and recording the subsequent aircraft motions, holding controls fixed. Time histories of motion following a vertical gust disturbance are presented in Figures 29 and 30 at airspeeds of 140 and 160 knots. At 140 knots, the effect of pitch/yaw coupling noted in the dynamic directional maneuvers contributes to an apparent pitch instability. With a right sideslip (left yaw), the pitch response appears more unstable than with a left sideslip. The same trend is apparent at 160 knots (Figure 30), where a pure pitching motion with the 2.2-degree stabilizer angle is stable in contrast to the unstable pitching motion with the 6.8-degree stabilizer setting. Very small roll and sideslip angles are developed at the 2.2-degree stabilizer setting, while large roll angles appear with the stabilizer at 6.8 degrees. These roll angles should result in a right sideslip, but since no significant sideslip angle was recorded, it is concluded that the helicopter motion included a yawing velocity to the left which compensated for the sideslip velocity and introduced the destabilizing pitch/yaw coupling.

At 170 knots (Figure 31), the aircraft response appears to be nearly pure pitch, and stability is positive. The effect of the initial longitudinal cyclic input is compensated for to some extent by over-shoot in returning the stick to trim, which results in modification of the response characteristics. Thus, the effect of wing lift on dynamic stability is not entirely clear from these tests, but comparison of the time histories shown in Figure 9 of Reference 2 suggests that the wing installation improves dynamic stability by reducing the destabilizing moment of the rotor with angle of attack.

INFLUENCE OF THE WING ON MANEUVER CAPABILITY

The wing installation provides an increase in the maximum normal load factor attainable during maneuvers which can be divided into two components. The portion of the total aircraft weight carried by the wing, in unaccelerated flight, increases the rotor load factor capability at a given airspeed. In addition, during the maneuver, the increasing wing lift further reduces the rotor load required to develop a given overall load factor. A graphical presentation of these effects is shown in Figure 6.

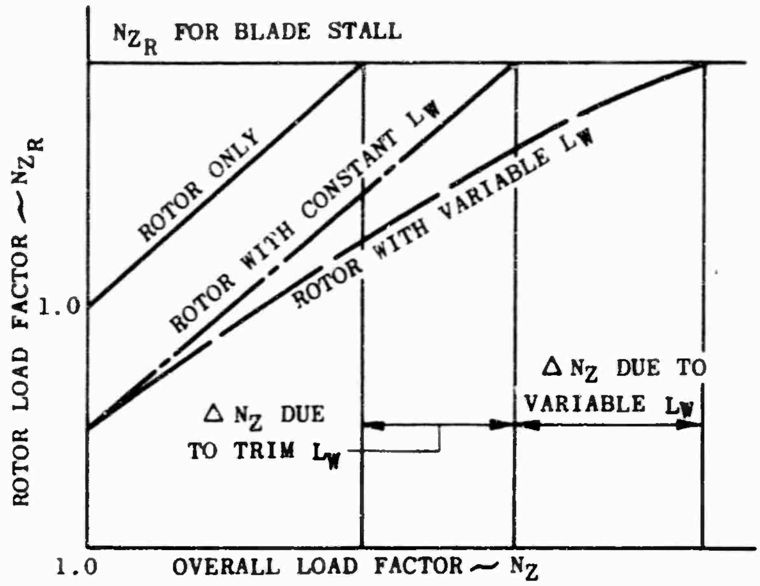

Figure 6. Schematic Diagram Showing the Effect of Wing Lift on Rotor Load Factor.

Figure 32 illustrates the variation of rotor load factor as a function of overall load factor in symmetrical pull-up maneuvers. At 160 knots it is evident that the wing carries an increasing portion of the total load as the overall load factor increases, thereby providing substantial rotor blade stall relief for maneuvering flight. A maximum overall load factor of 2.04 g was developed at 160 knots as shown in Figure 32, which is well above the 1.86 g established as a rotor stall limit load factor prior to the wing installation. During this maneuver a peak load factor of 1.14 g was developed by the rotor.

The significance of the results obtained in this phase of testing is that proper design of the compound helicopter can result in a vehicle with maneuver capability limited only by structural considerations. This will not be necessarily accomplished by the installation of the wing alone. Details which influence the shape of the rotor load factor curve as a function of overall load factor must also be considered. These include such items as the rate and magnitude of control input, the ratio of pitch response of the rotor and fuselage, and the wing size and aspect ratio.

Figure 33 presents the results of rotor load factor alleviation in banked turns to the right. Because this maneuver involves principally roll and yaw motions rather than pitch, it would be expected that the wing load factor change would be small with increasing overall load factor. The data indicate an increase, however, which is essentially linear with overall factor in contrast to the characteristic seen during symmetrical pull-up maneuvers where wing lift increases rapidly once the aircraft starts to pitch.

It is pointed out that the turns were accomplished at a fixed horizontal stabilizer incidence; consequently, they represent a configuration with main rotor control only. The rate of change of rotor load factor with overall load factor would be considerably modified using combined rotor and aerodynamic surface control. This is apparent from examination of Figures 33 and 34 at 140 knots, for example. For a 4.6-degree equivalent up-horizontal stabilizer movement, the load factor on the wing is increased by over 40 percent at an overall load factor of 1.6. This result suggests that rotor stall margins can be increased during turn maneuvers by the use of an elevator to load the wing.

STRUCTURAL LOADS AND VIBRATION

The effect of wing lift on critical structural loads and aircraft vibration is presented in Figures 35 through 47.

Main Rotor Loads

Wing lift relieves main rotor and servo-flap vibratory bending moments. This effect has been noted in previous UH-2 test data where a decrease in gross weight (rotor lift) results in reduction of both steady and vibratory rotor loads. The data presented in Figures 35 and 36 indicate a significant reduction in the bending moments at critical blade and servo-flap locations when compared with the data from Reference 1 at approximately the same level of thrust augmentation. This is particularly evident in the case of flapwise bending at station 190, where an increase of approximately 20 knots was realized for the same bending moment level.

Main rotor loads will present no problem with further expansion of the airspeed envelope. The addition of increased thrust augmentation, necessary to attain higher airspeeds, will afford a reduction in the loads, as demonstrated during the jet-augmented program reported in Reference 1.

Control system loads monitored during the program showed consistently low levels which were unaffected by changes in either airspeed, thrust augmentation, or wing lift.

Tail Rotor Loads

Tail rotor stresses, shown in Figure 37, were relatively unchanged from the levels previously noted for the jet-augmented helicopter and remained well below the endurance limit for all conditions flown.

Airframe Loads

The reduction of main rotor bending moments noted as a function of wing lift is reflected in airframe loads examined at selected locations on the aircraft. A substantial reduction of the vibratory stresses was noted for the main rotor transmission mount support tubes with moderate reductions seen in the other areas. Airframe structural loads were found to be within acceptable limits for all conditions examined. The trends established for critical airframe loads are summarized in Figures 38 through 42.

Vibration

The vibration levels of the compound helicopter are summarized in Figures 43 through 47.

The pilot's station and station 50 fuselage vibrations, shown in Figures 43 and 44, are relatively unchanged from the levels previously attained on the jet-augmented helicopter for comparable airspeeds and levels of thrust augmentation.

A slight decrease in the vertical accelerations is noted at the higher airspeeds for the center of gravity, with a somewhat larger influence noted for station 400.

In contrast, the main rotor transmission lateral accelerations, shown in Figure 47, are substantially reduced compared to the data presented in Figure I-25 of Reference 1. This reflects the reduction noted in airframe loads with wing lift.

REDUCED POWER DESCENT AND AUTOROTATION

In reduced power descents and full autorotation, handling qualities were found to be satisfactory provided the wing angle of attack did not reach the stall value. During this testing it was found that the wing operates at or near the maximum lift coefficient, and wing lifts in excess of 2000 pounds were recorded at airspeeds between 80 and 90 knots. When extensive wing flow separation occurred, shown typically in Figure 48, erratic rolling moments resulted. Adequate roll control was available, however, and it was comparatively easy to reduce the wing angle of attack below the stall angle by adjustment of the pitch attitude of the helicopter by use of the variable incidence horizontal stabilizer.

Entry into autorotation was accomplished with the jet engine off by lowering the collective pitch followed by T-58 throttle reduc-

tion by the pilot. This procedure is similar to the normal entry into autorotation with a pure helicopter and resulted in satisfactory controllable descents, although the rotor speed tended to drop below the normal minimum of 92 percent by as much as 10 percent because of the reduced rotor load. This result was anticipated, and to increase the power-off rotor speed the blade servo-flaps were rigged to produce a lower blade pitch angle with full-down collective stick. This procedure resulted in recovery of approximately 6 percent of the initial 10 percent loss, which provided sufficient rotor energy to permit a flare maneuver in a fule autorotation landing.

POWER LOSS

Preliminary investigation of the effect of a T-58 po. .r failure was made at airspeeds of between 50 and 115 knots as part of the investigation of steady-state autorotation. This testing, which was accomplished with the jet engine off, showed the developemnt of high wing lift, which would tend to aggravate the rapid decay of rotor speed observed with the jet-augmented configuration. Reduced rotor speed results in decreased rotor control power, which introduces difficulty in counteracting the rolling moments associated with wing stall. It became apparent that further evaluation of T-58 power loss would require an extensive exploration outside the scope of this program.

Sudden reduction of YJ-35 power was accomplished at a true airspeed of 166 knots for comparison with the maneuver performed with the jet-augmented helicopter. The results obtained at this speed are presented in Figure 49. The major effect noted by the pilot was a rapid decrease in airspeed similar to that shown for the jet-augmented configuration. Aircraft response was docile with reference to all other axes, and it is, therefore, concluded that the presence of the wing had no significant effect. No serious problems would be anticipated at the power-limited airspeed for this aircraft.

LOW SPEED MANEUVER

Pilot evaluation of low speed maneuvers close to the ground, including hover, sideward flight, and rearward flight, indicates some adverse effects due to the presence of the wing. While normal air taxi maneuvers can be performed at wheel heights of 10 to 20 feet, the pilot's visual reference to the ground is degraded and turbulent flow along the ground underneath the wing results in small but erratic displacements about all the helicopter axes which require constant corrective control inputs. In the final stages of the anding maneuver as the normal ground cushion builds up, the turbulent flow beneath the wing becomes more pronounced.

TABLE II

COMPARISON OF ACTUAL ADVANCING TIP MACH NUMBER WITH PREDICTED VALUES

L_R/N_R^2 lb. σ'	μ	MRHP/ $N^3\sigma'$	M_T Crit Predicted	M_T Crit Measured
6260	0.459	925	0.866	0.865
6310	0.458	1010	0.865	0.875
8480	0.475	952	0.856	0.861
8290	0.450	823	0.856	0.875
6960	0.475	845	0.867	0.862
8540	0.485	955	0.855	0.866
8290	0.449	771	0.856	0.862
8430	0.451	856	0.853	0.864
8250	0.435	830	0.853	0.862
7620	0.450	720	0.862	0.867
7740	0.449	1185	0.846	0.846
9580	0.449	950	0.845	0.844
9180	0.423	824	0.846	0.853
9580	0.366	714	0.839	0.831

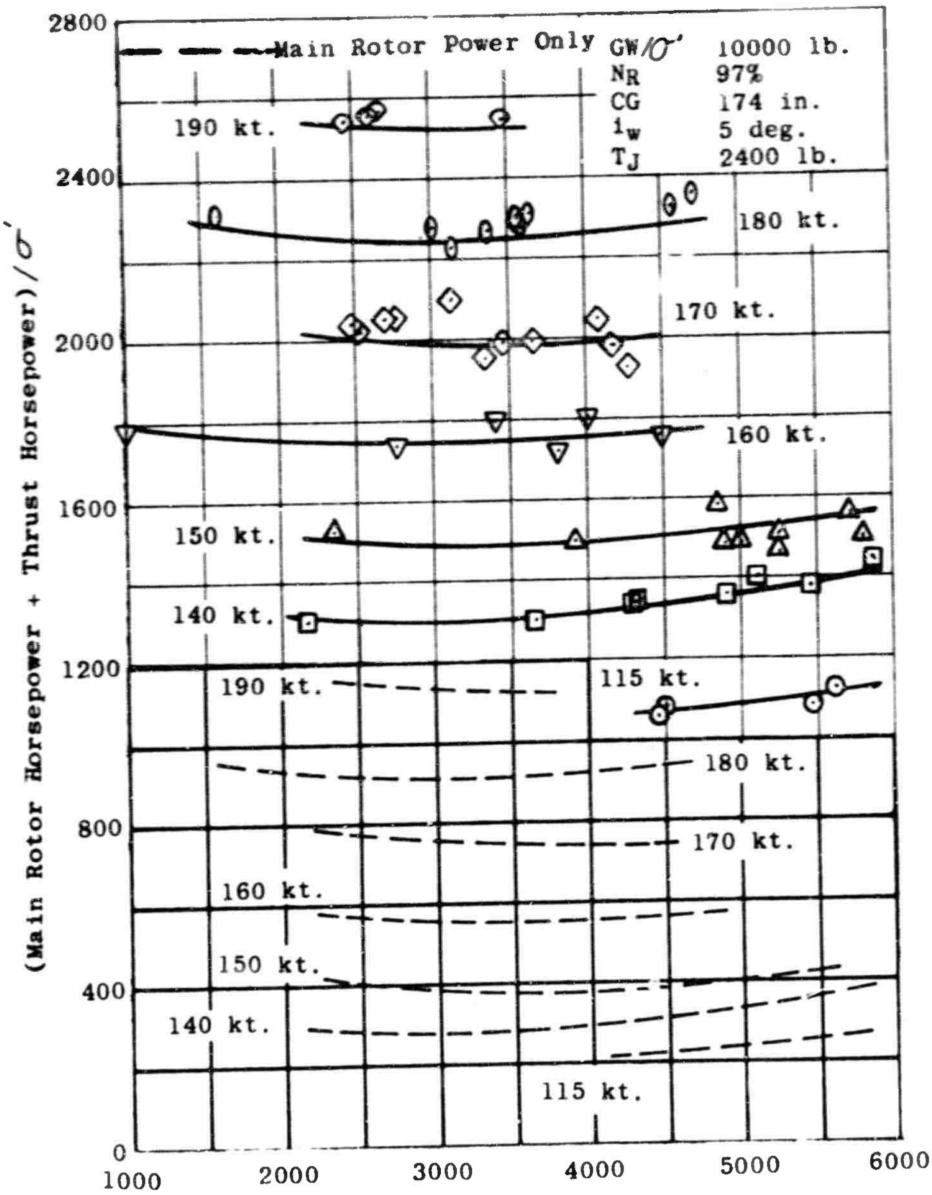

Figure 7. Horsepower Required for Level Flight as Affected by Wing Lift.

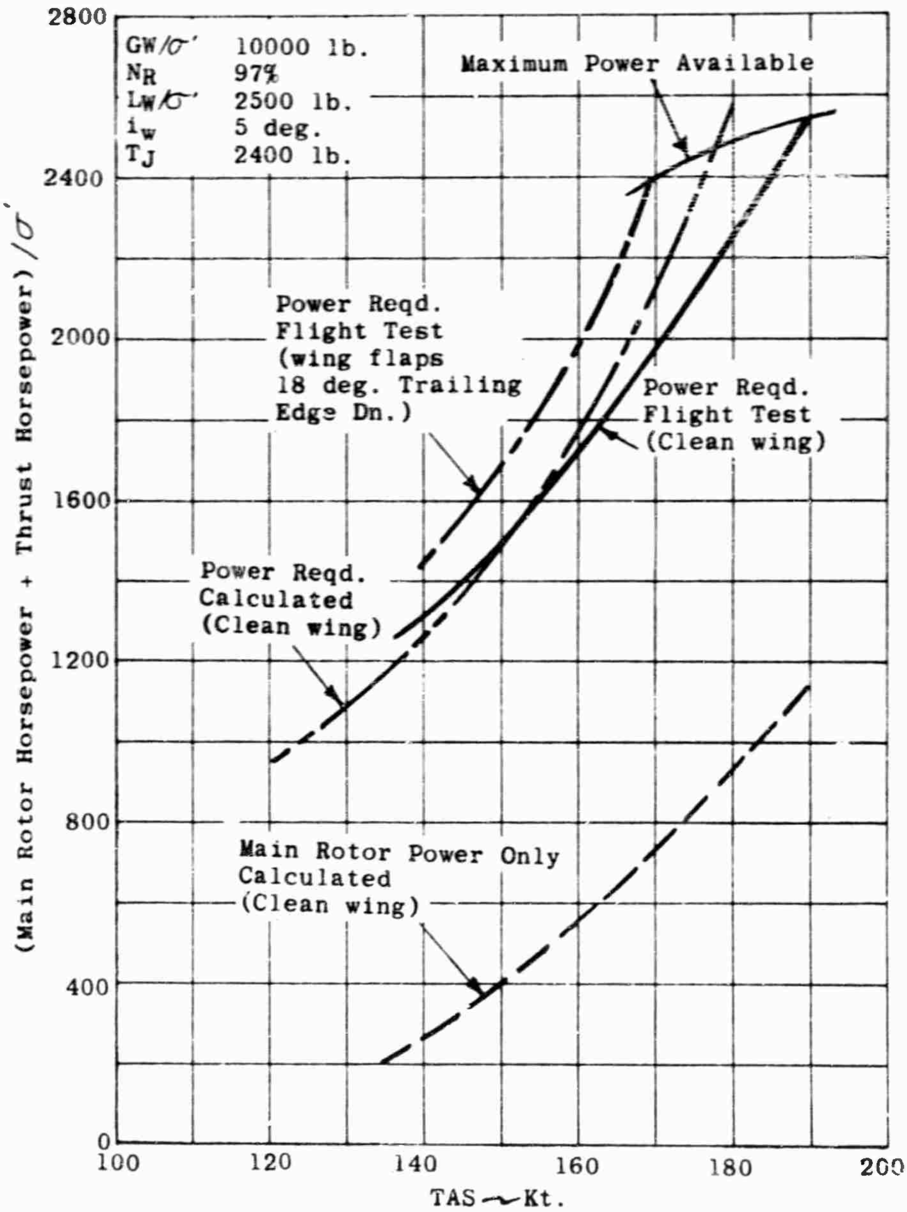

Figure 8. Theoretical and Experimental Correlation of Horsepower Required for Level Flight.

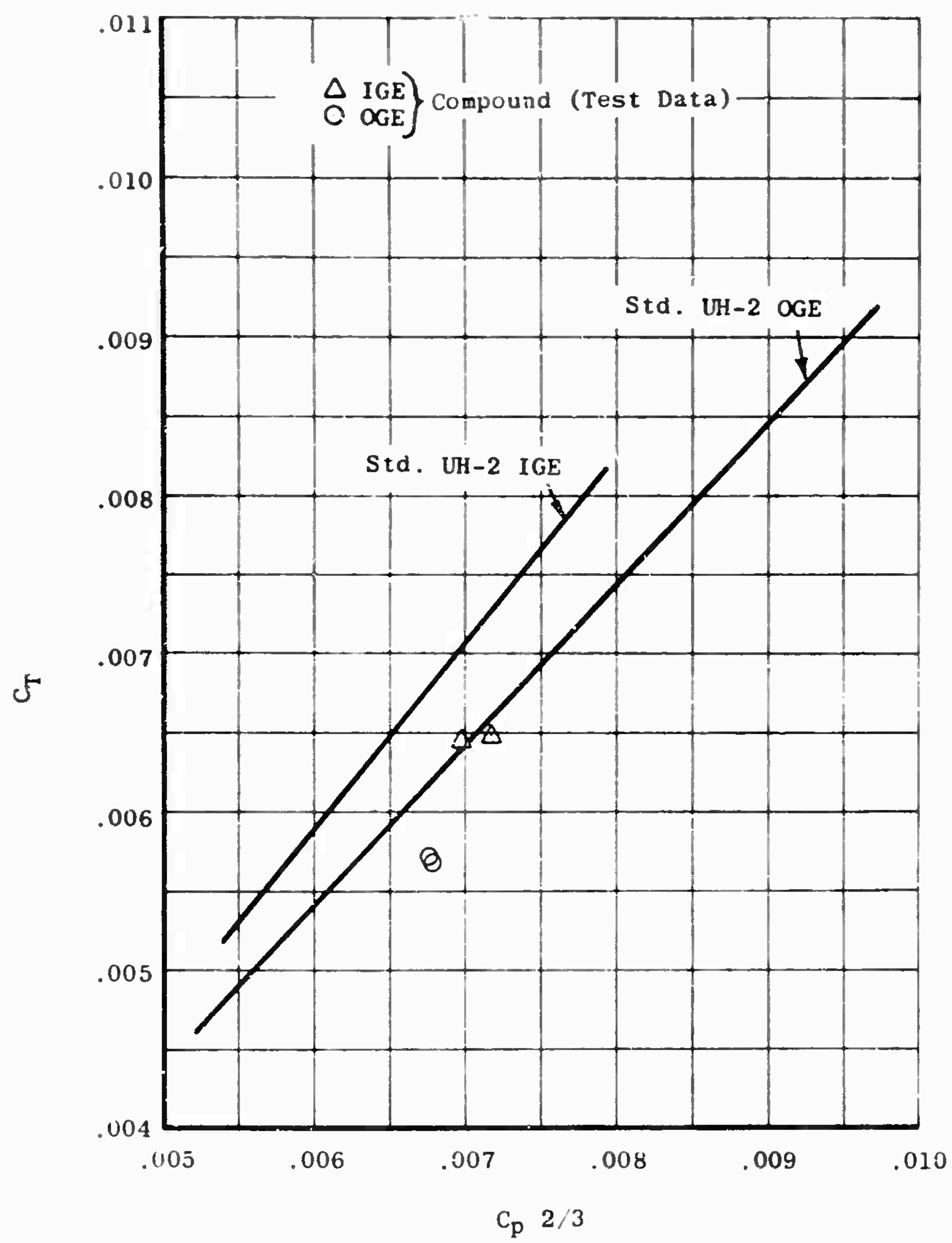

Figure 9. Comparison of Compound and Standard UH-2 Hover Performance.

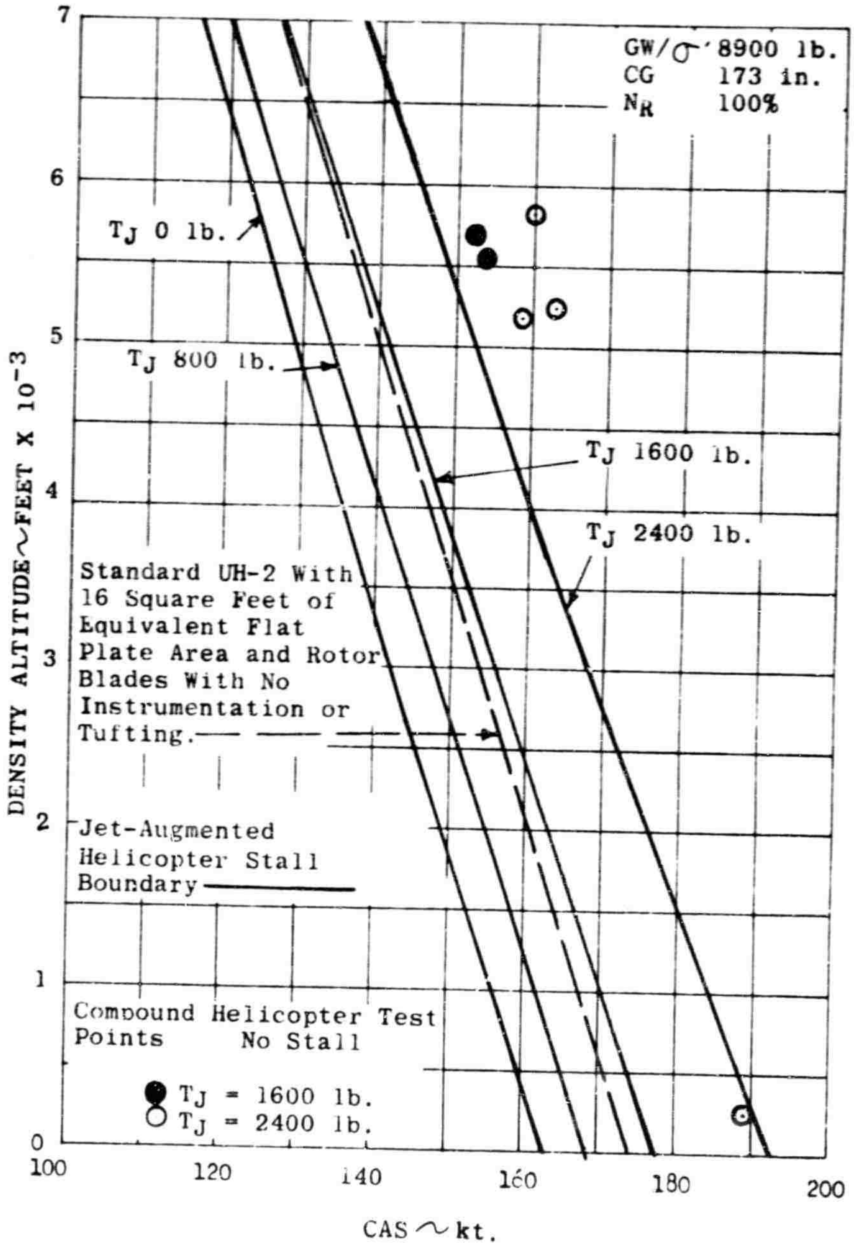

Figure 10. Penetration by the UH-2 Compound Beyond Stall Boundaries Established by the Jet-Augmented Helicopter.

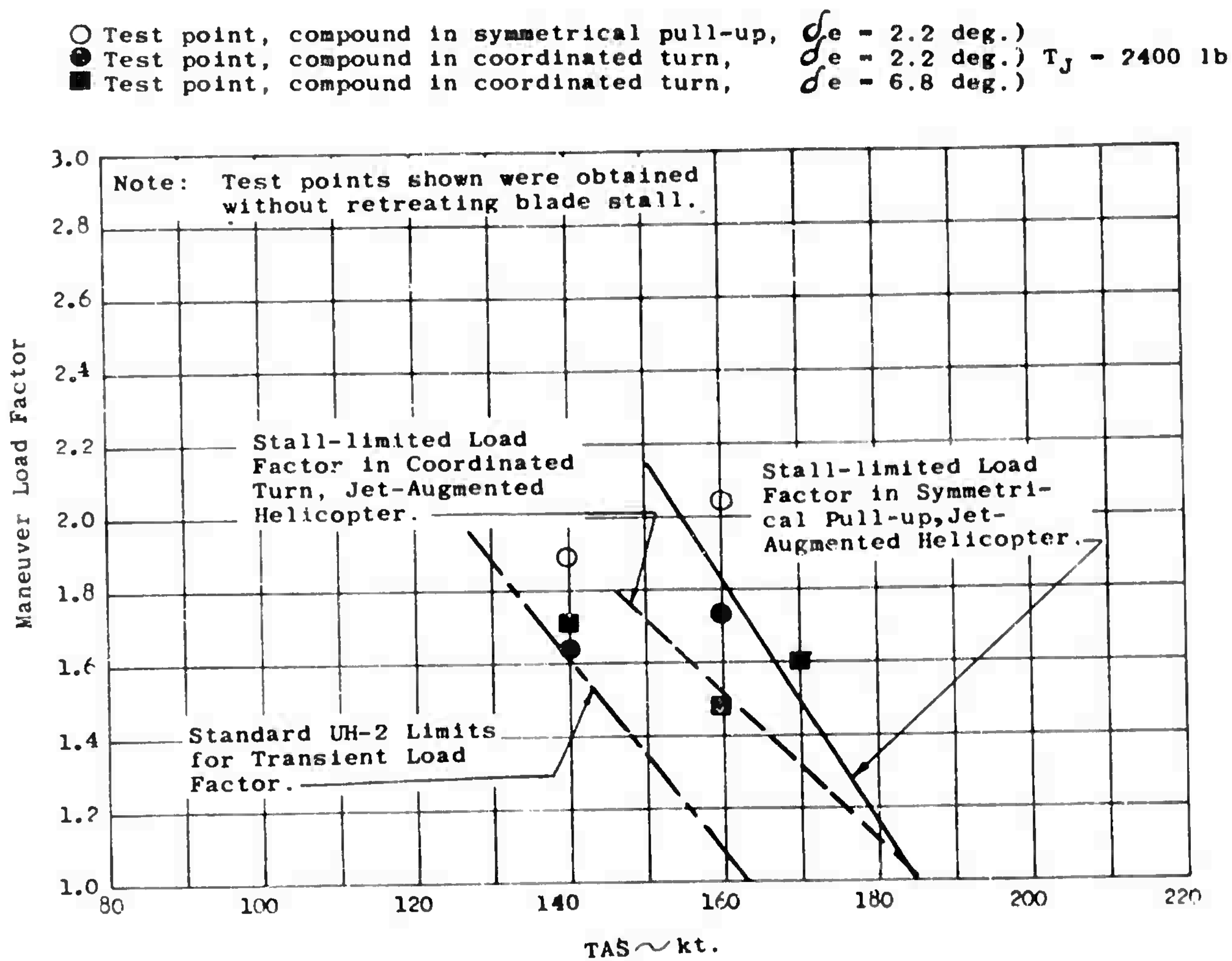

Figure 11. Maneuver Load Factor Achieved on the Compound With No Retreating Blade Stall.

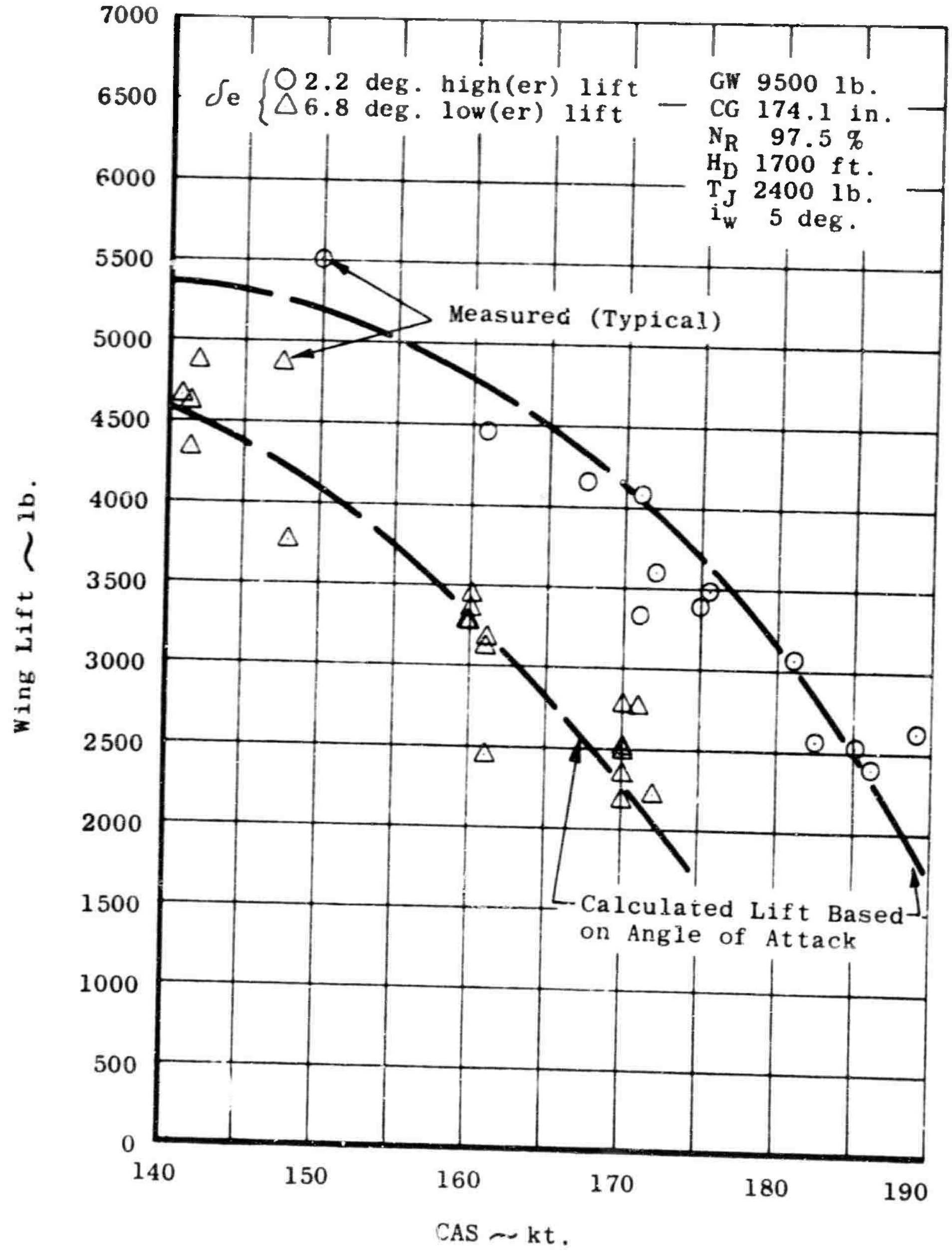

Figure 12. Wing Lift as Affected by Airspeed and Horizontal Stabilizer Incidence.

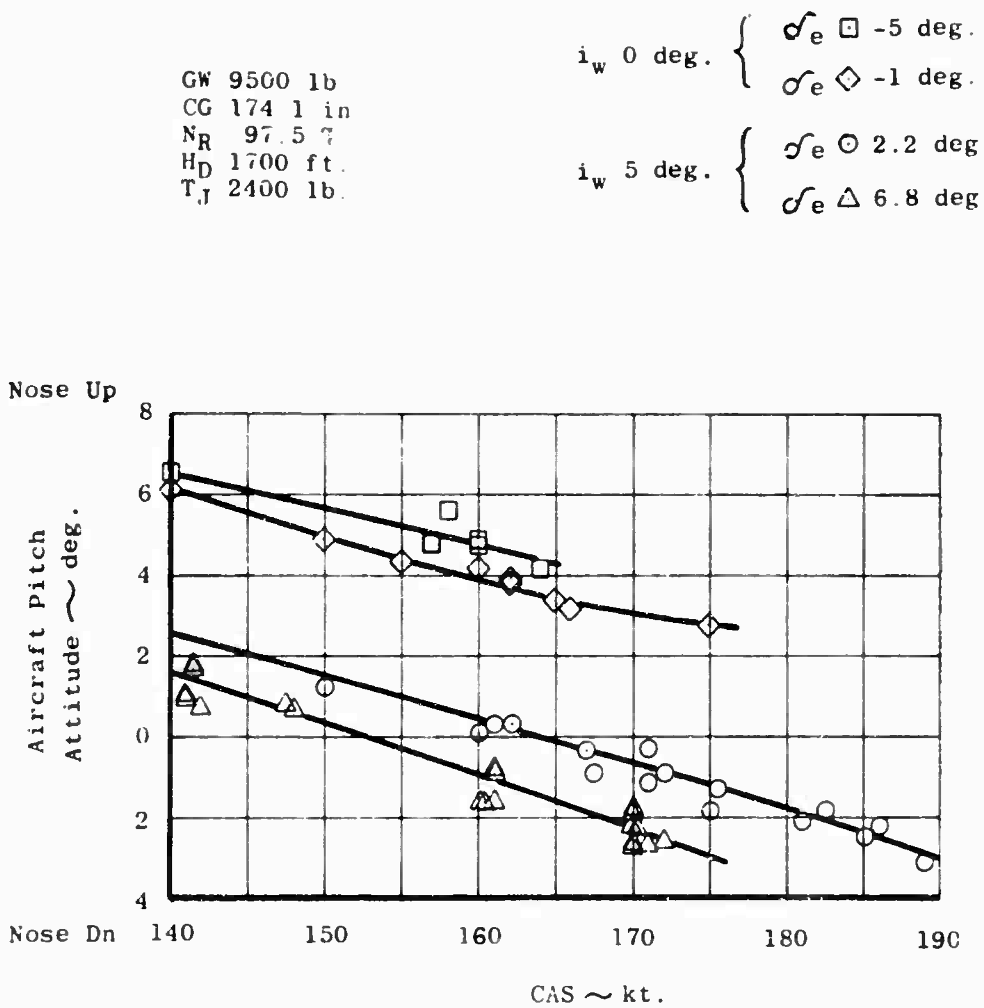

Figure 13. Aircraft Pitch Attitude as Affected by Horizontal Stabilizer Incidence.

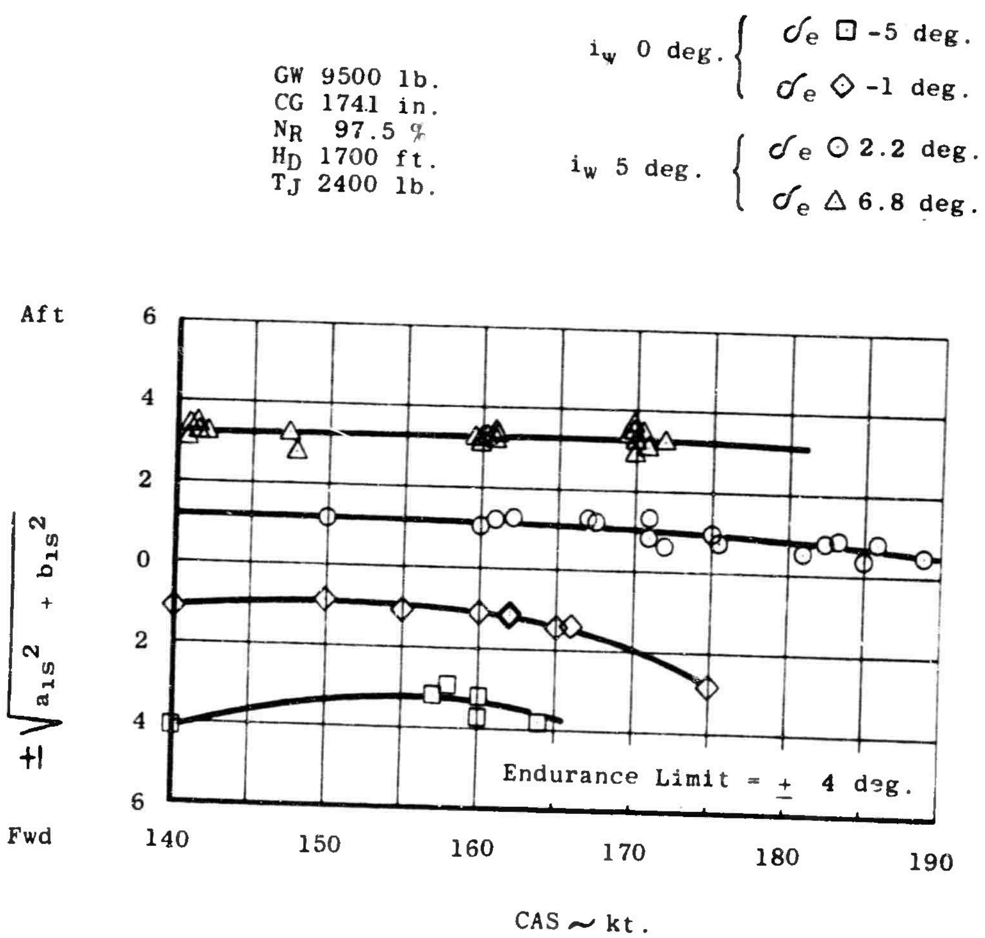

Figure 14. Main Rotor Flapping Angle Relative to the Shaft as Affected by Horizontal Stabilizer Incidence.

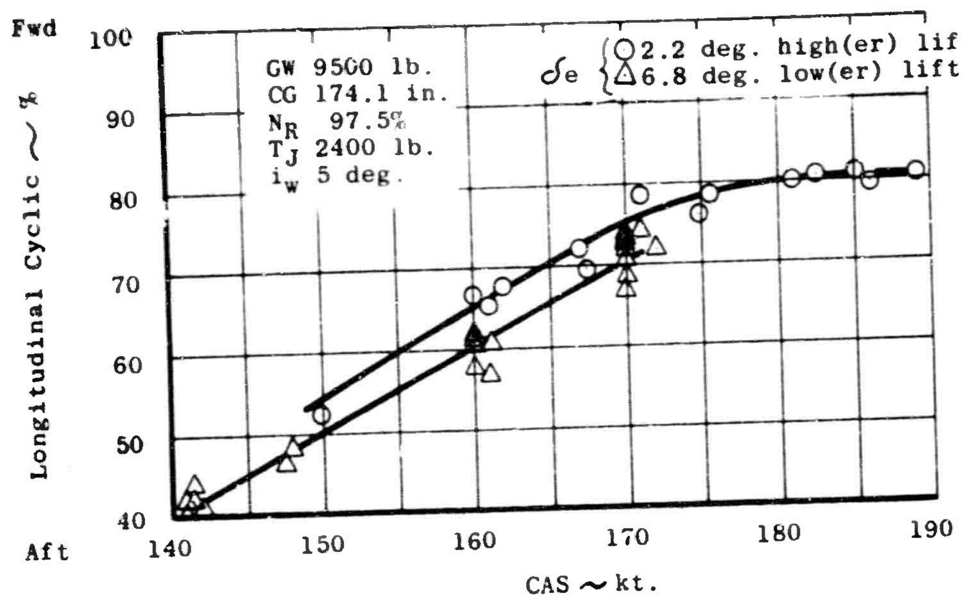

Figure 15. Longitudinal Cyclic Variation With Airspeed as Affected by Horizontal Stabilizer Incidence.

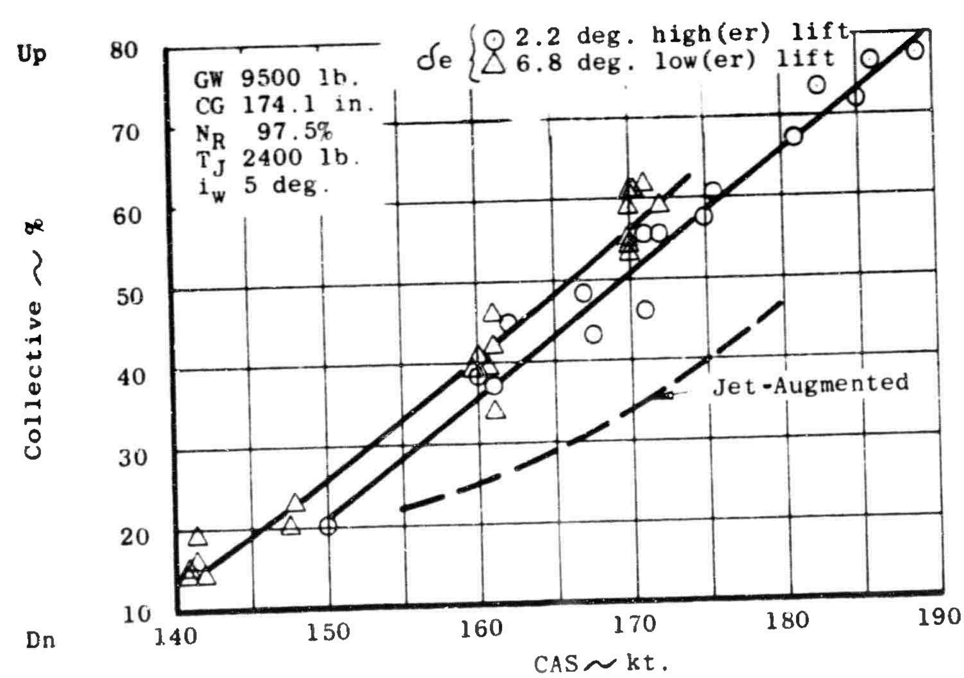

Figure 16. Collective Variation With Airspeed as Affected by Horizontal Stabilizer Incidence.

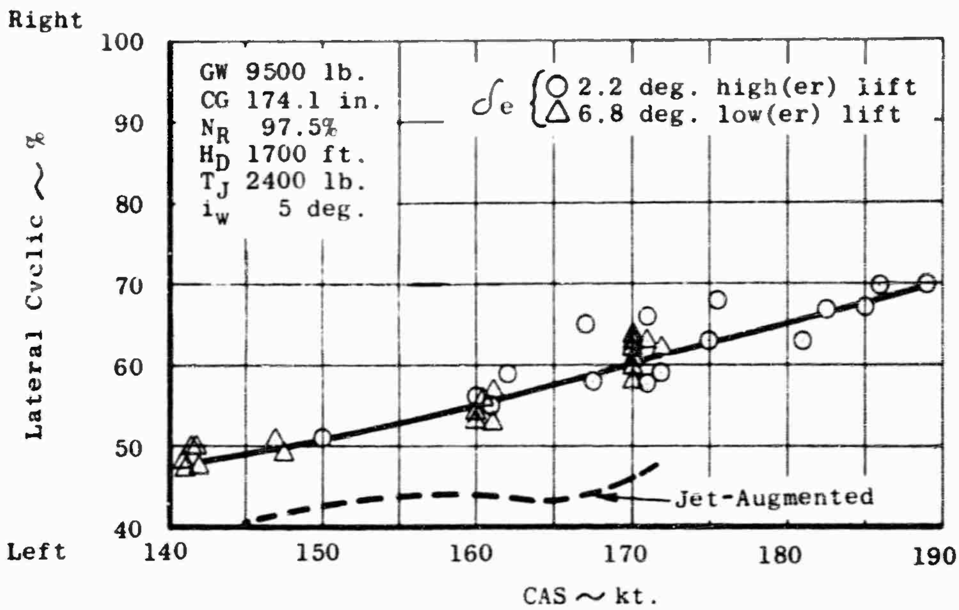

Figure 17. Lateral Cyclic Variation With Airspeed as Affected by Horizontal Stabilizer Incidence.

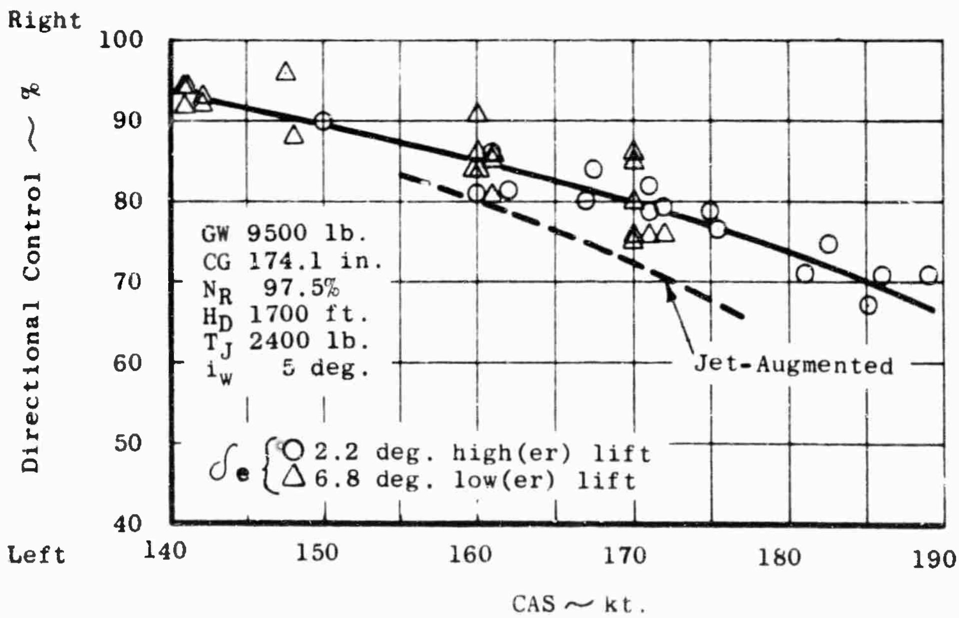

Figure 18. Directional Control Displacement With Airspeed as Affected by Horizontal Stabilizer Incidence.

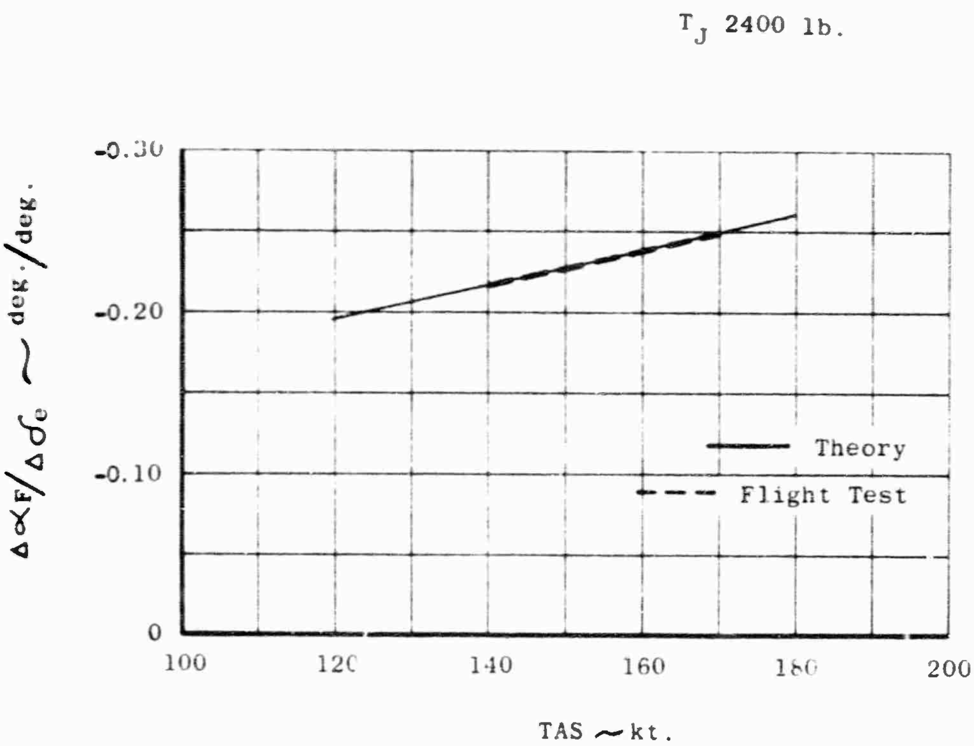

Figure 19. Theoretical and Experimental Correlation of Fuselage Attitude per Degree of Horizontal Stabilizer Incidence Angle.

T_J 2400 lb.

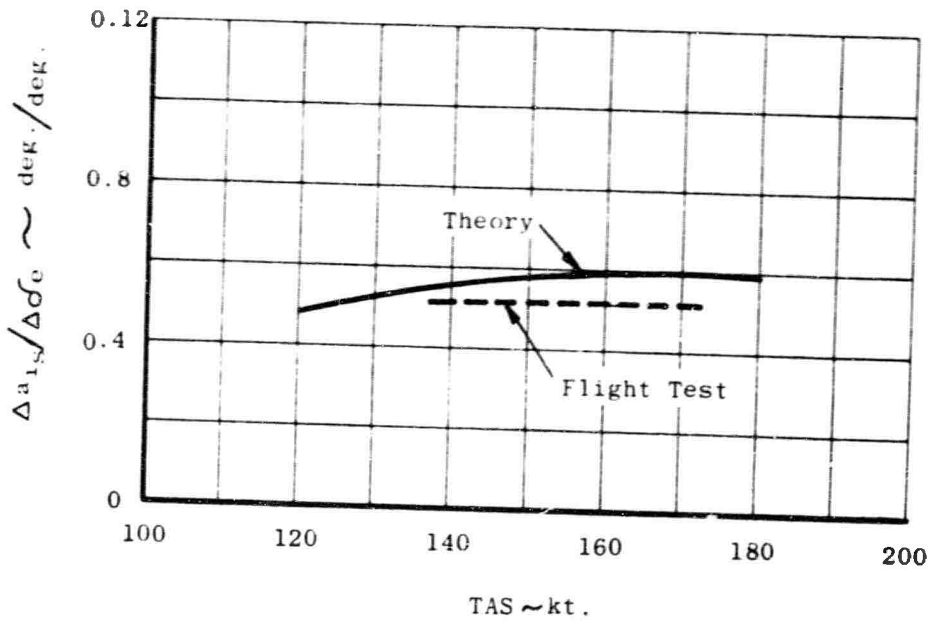

Figure 20. Theoretical and Experimental Correlation of Flapping (a_{1S}) per Degree of **Horizontal** Stabilizer Incidence Angle.

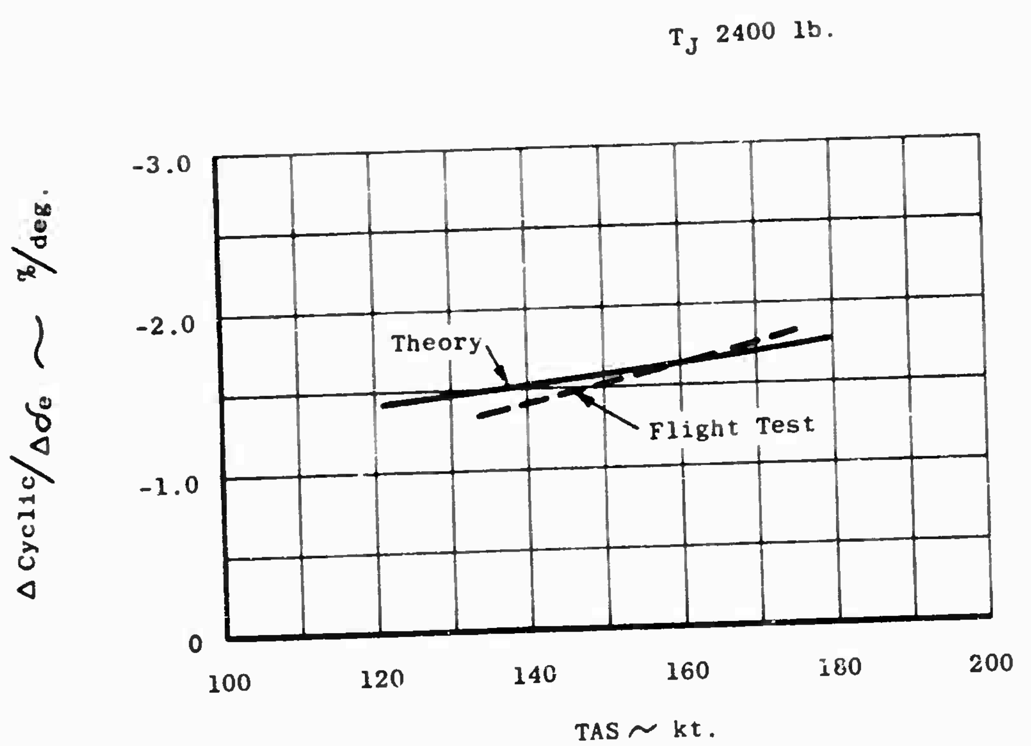

Figure 21. Theoretical and Experimental Correlation of Longitudinal Cyclic Variation per Degree of Horizontal Stabilizer Incidence Angle.

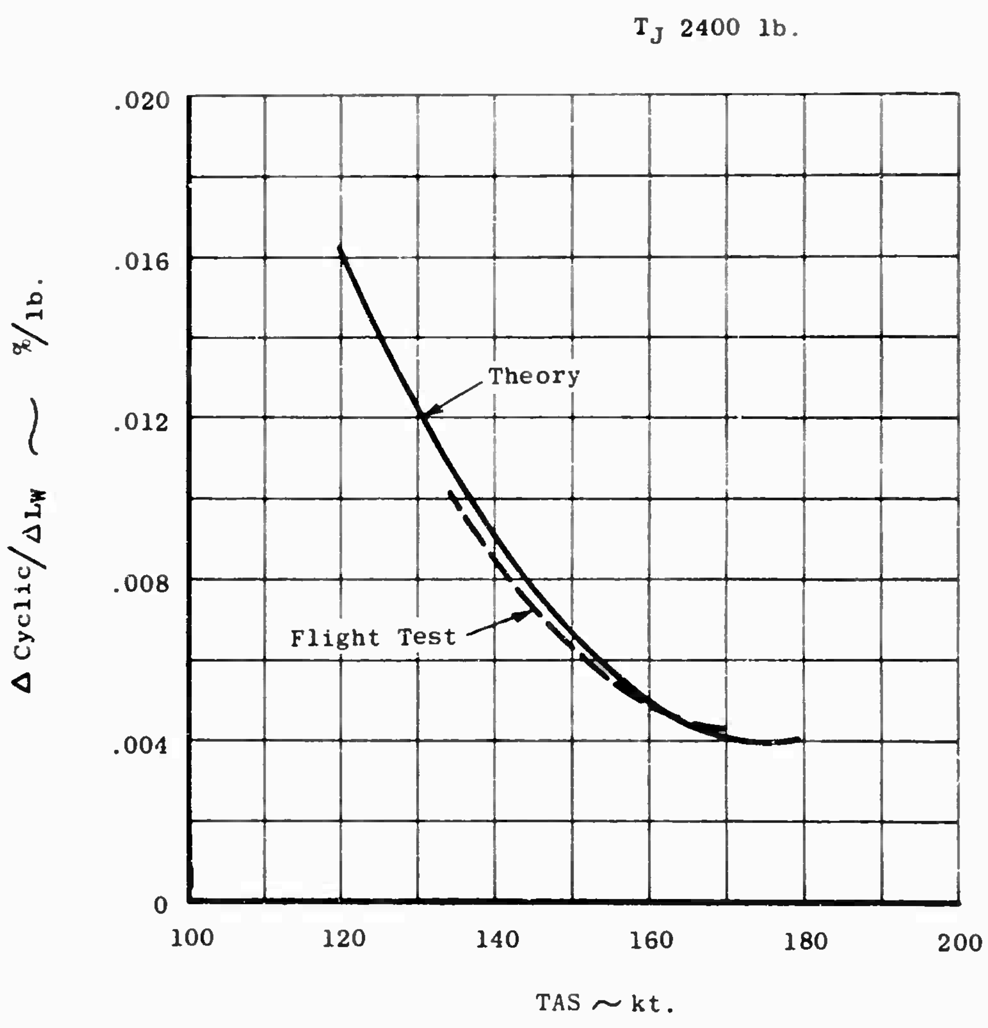

Figure 22. Theoretical and Experimental Correlation of Longitudinal Cyclic Variation per Pound of Wing Lift.

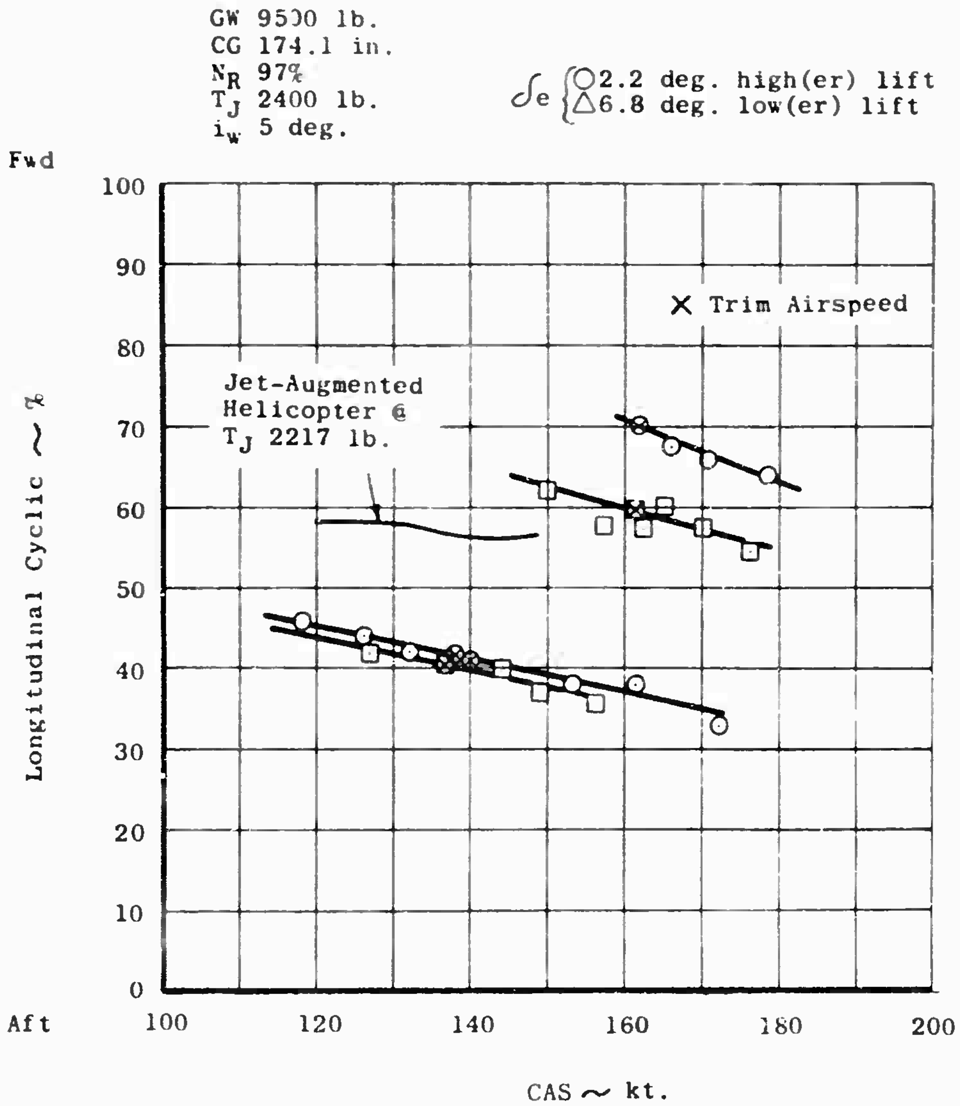

Figure 23. Effect of Wing Lift on Longitudinal Static Stability with Respect to Airspeed.

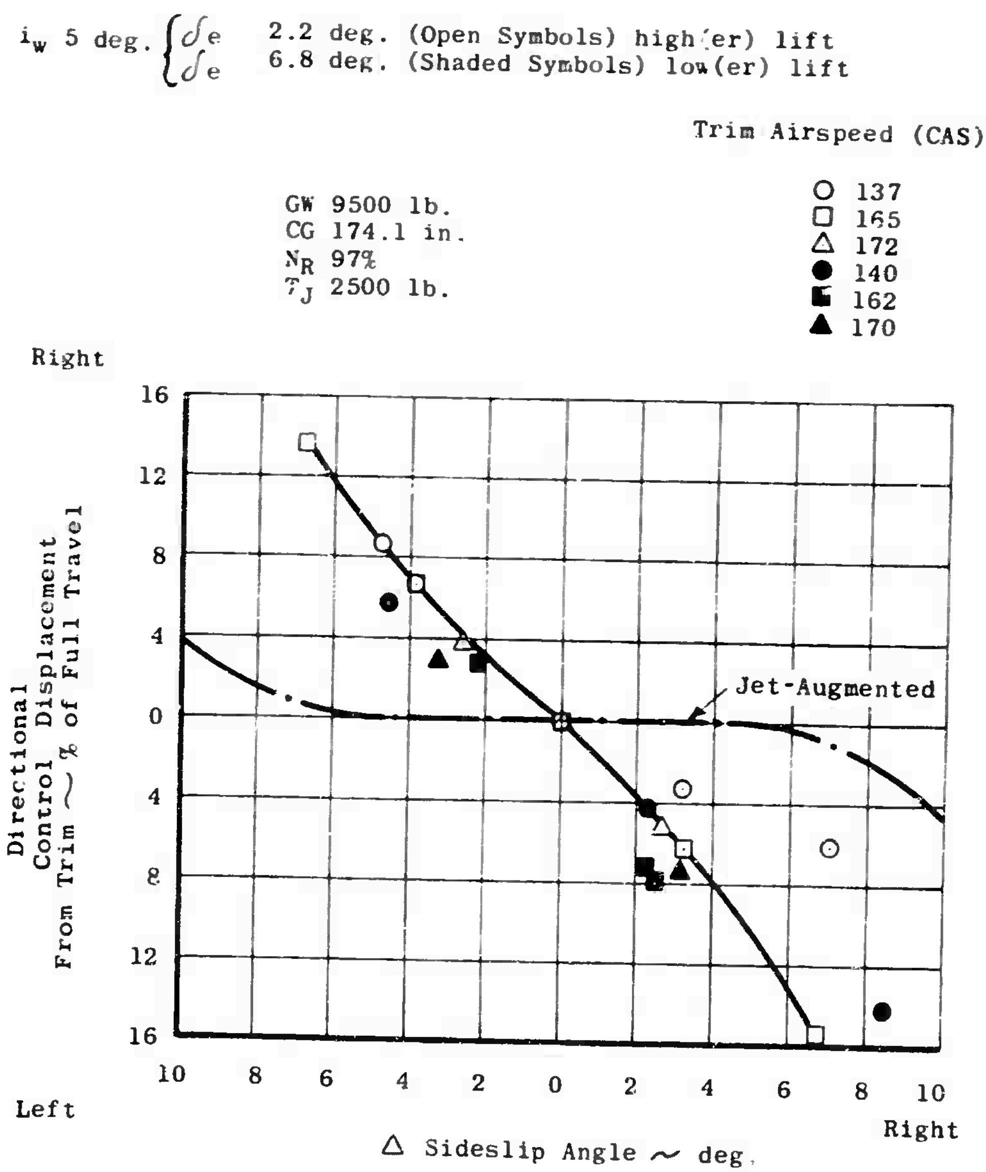

Figure 24. Effect of Wing Lift on Directi l Control Displacement as a Fun n of Sideslip Angle.

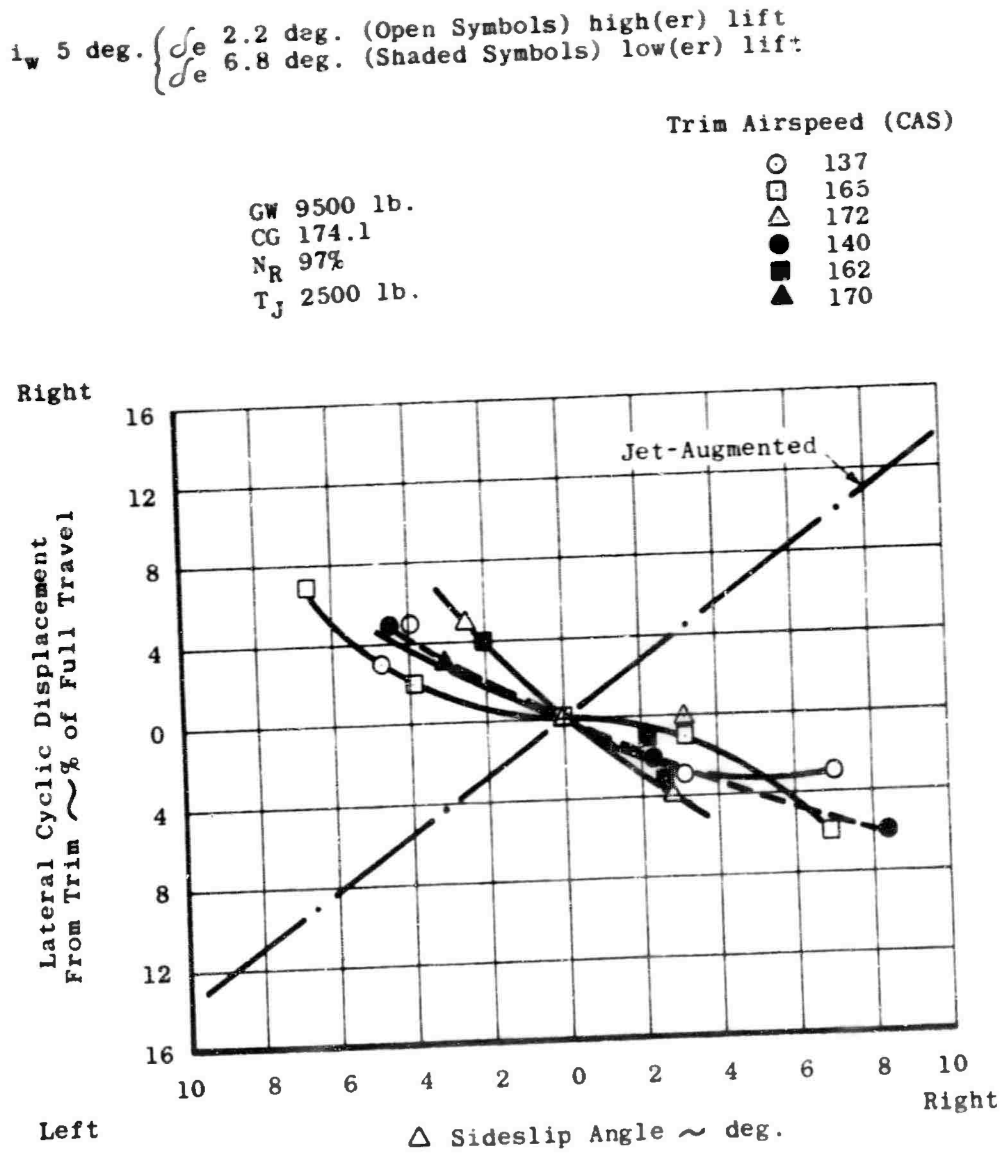

Figure 25. Effect of Wing Lift on Lateral Cyclic Control as a Function of Sideslip Angle.

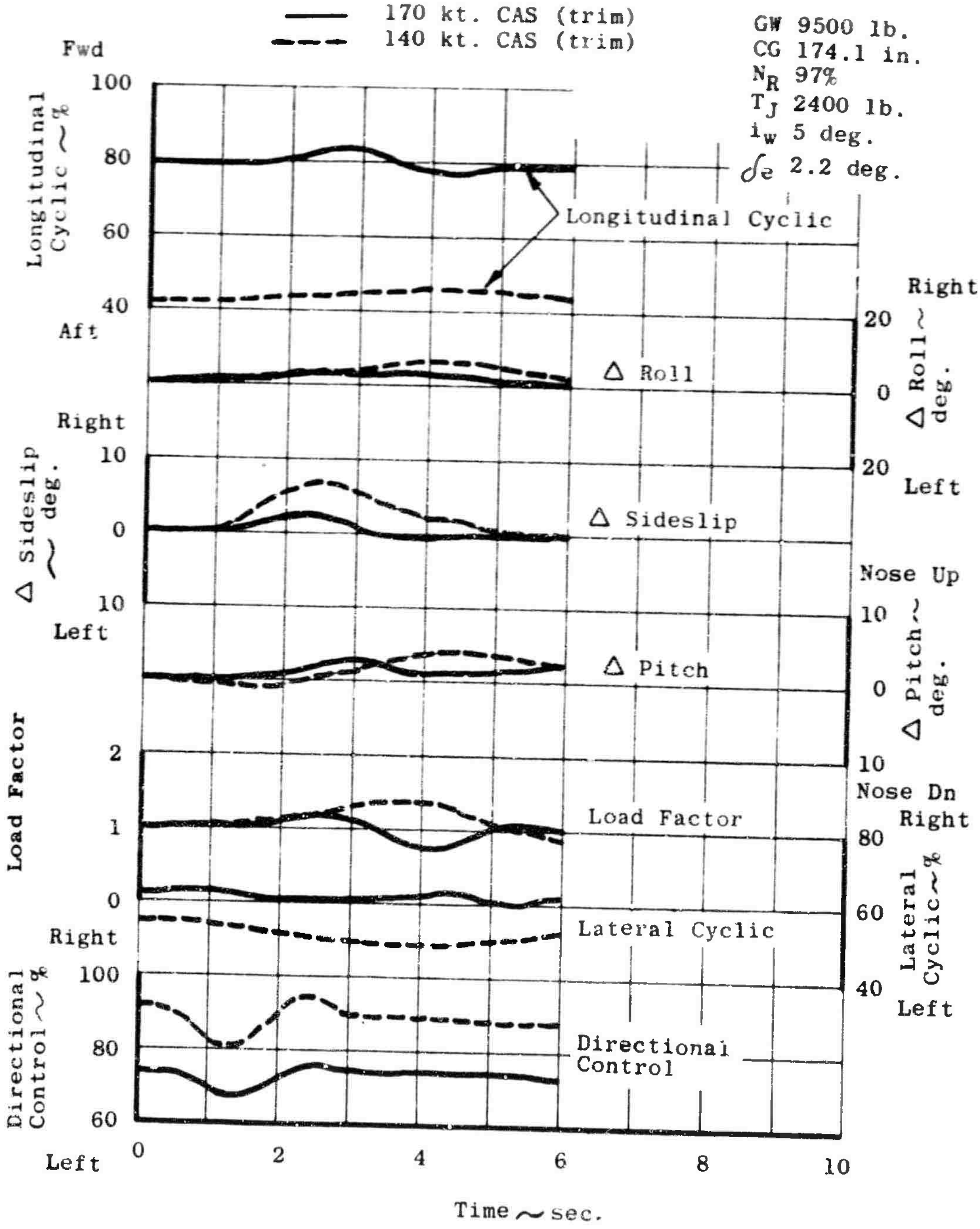

Figure 26. Effect of Airspeed on Helicopter Response to a Simulated Side Gust From the Right.

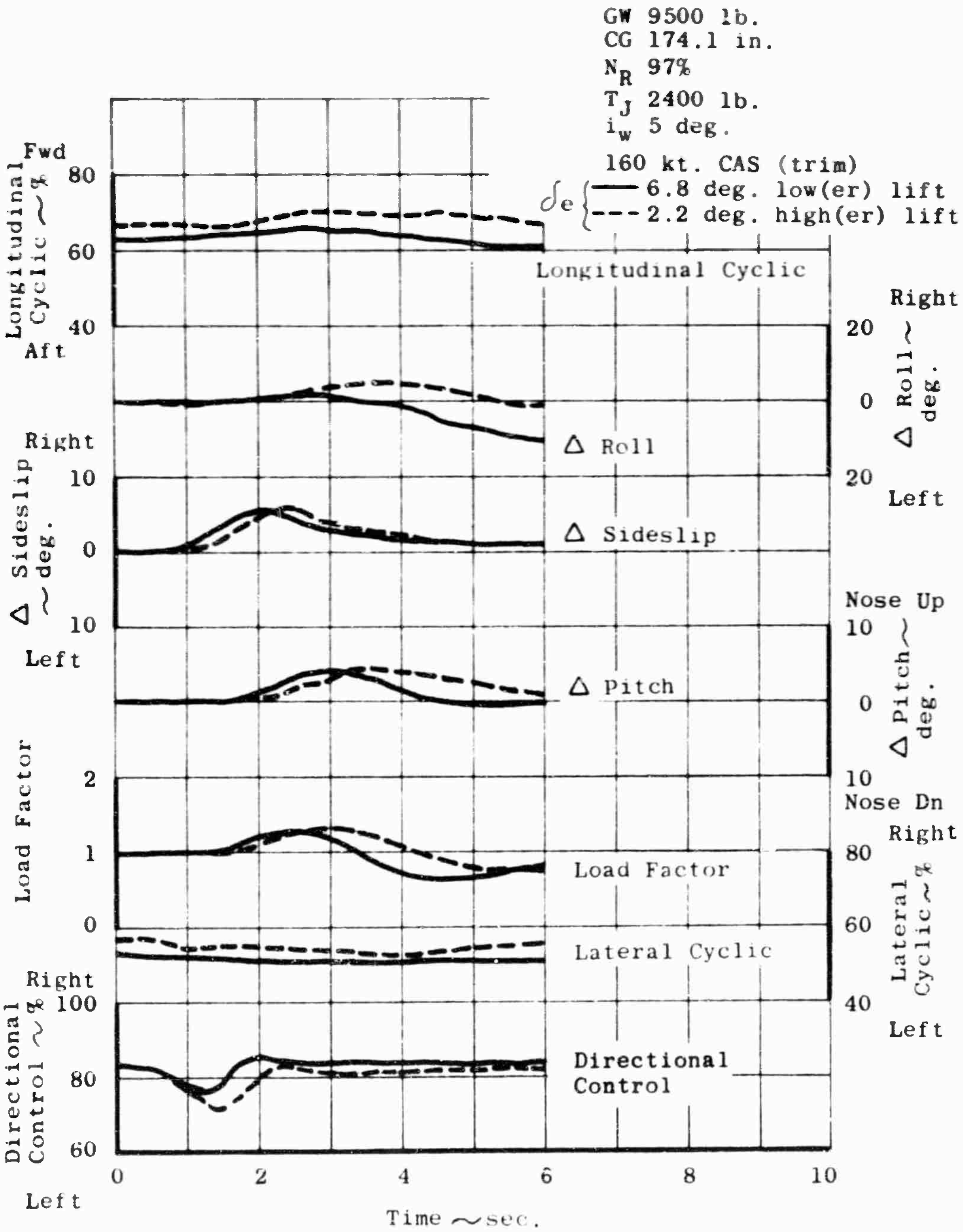

Figure 27. Effect of Wing Lift On Helicopter Response to a Simulated Side Gust From the Right.

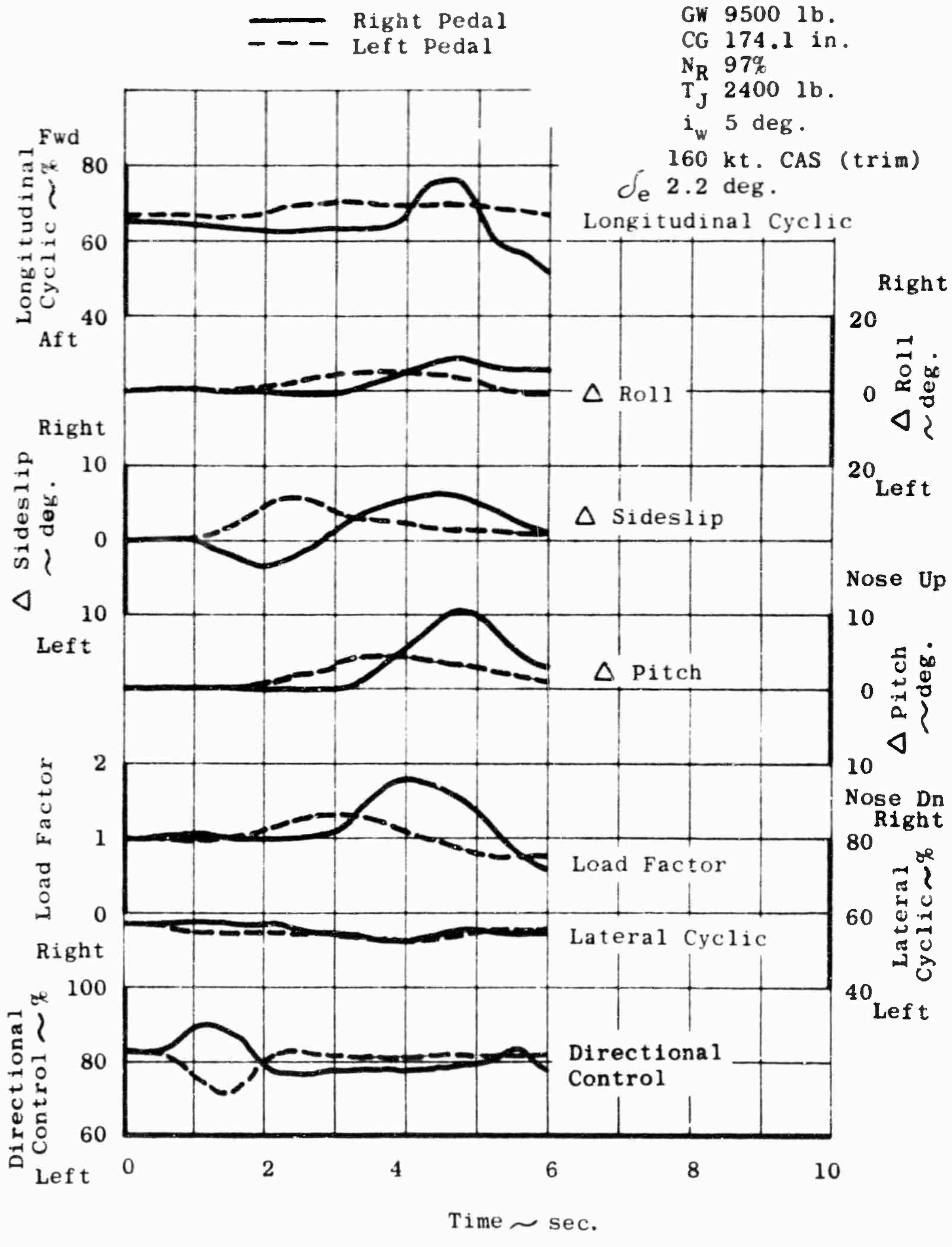

Figure 28. Helicopter Response to a Change in the Direction of Simulated Side Gusts.

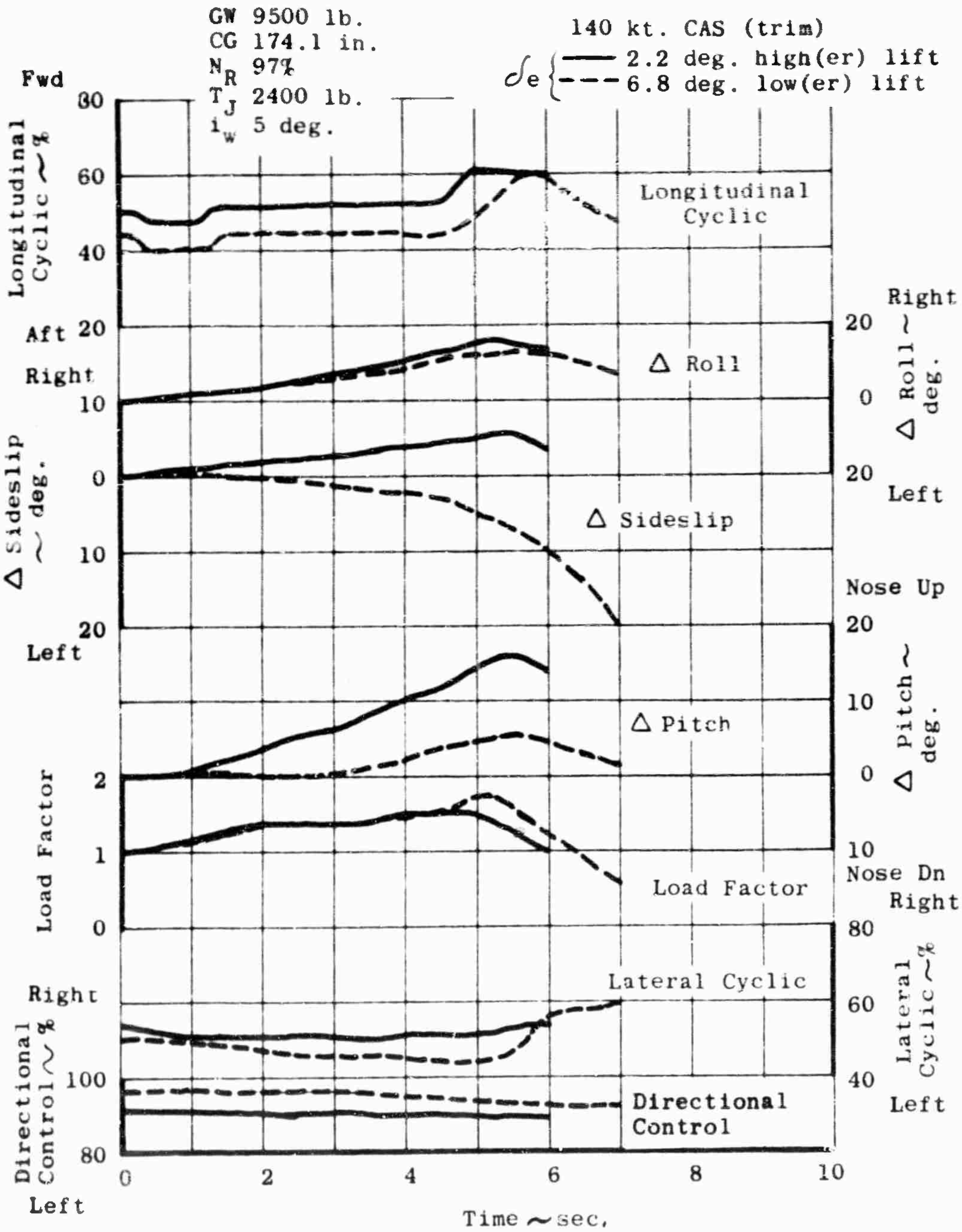

Figure 29. Helicopter Response to a Simulated Vertical Up-Gust as Affected by Wing Lift, CAS - 140 knots.

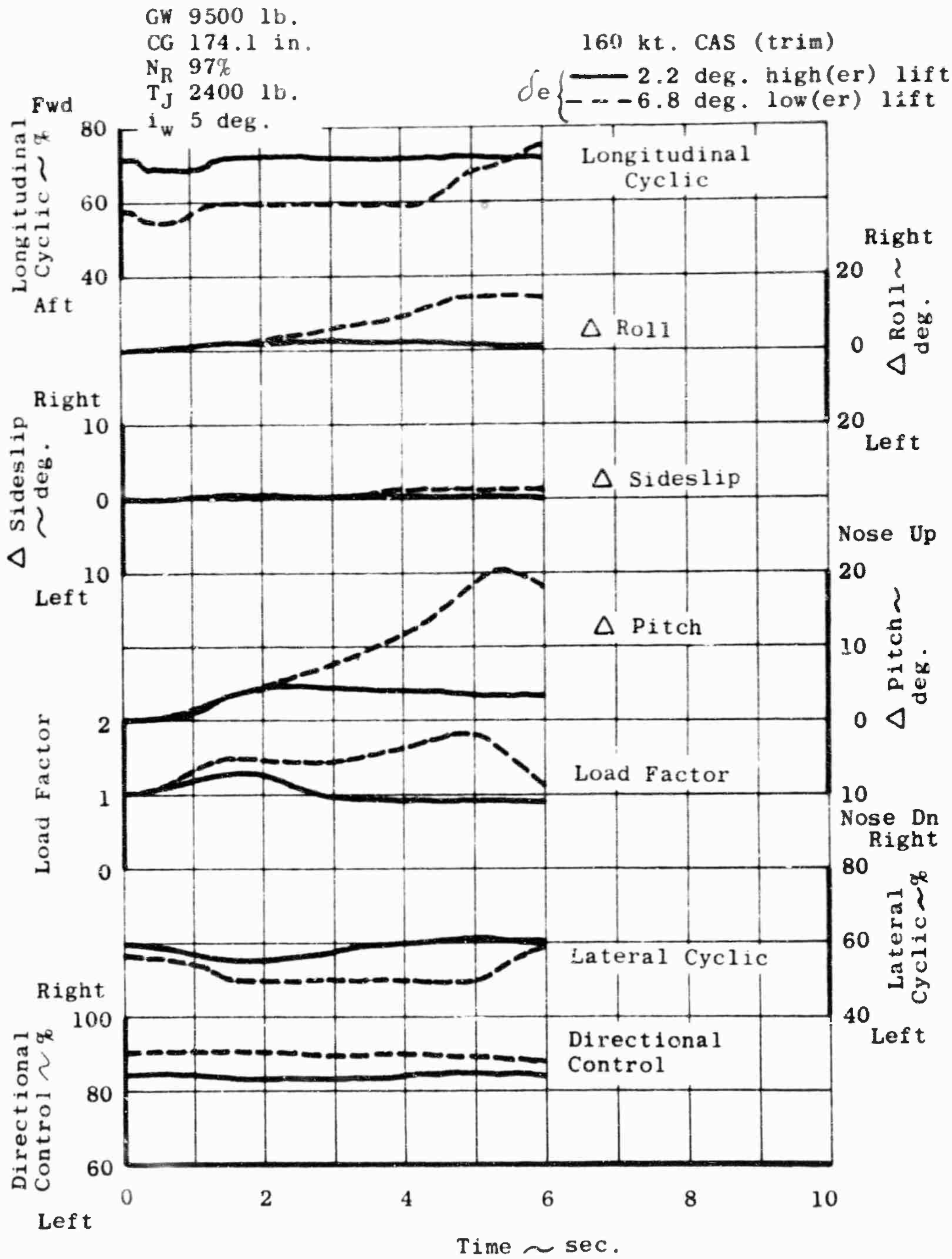

Figure 30. Helicopter Response to a Simulated Vertical Up-Gust as Affected by Wing Lift, CAS - 160 knots.

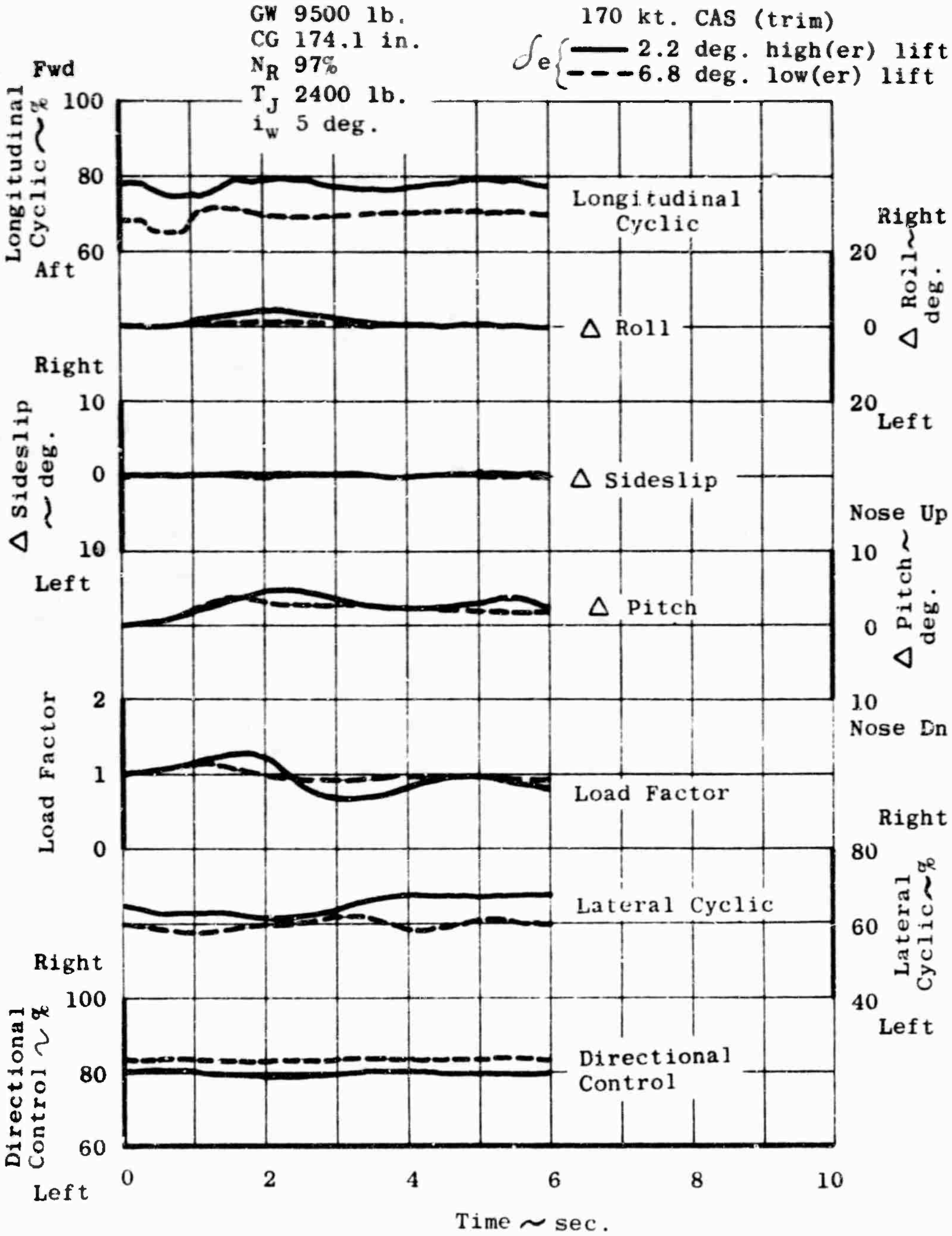

Figure 31. Helicopter Response to a Simulated Vertical Up-Gust as Affected by Wing Lift, CAS - 170 knots.

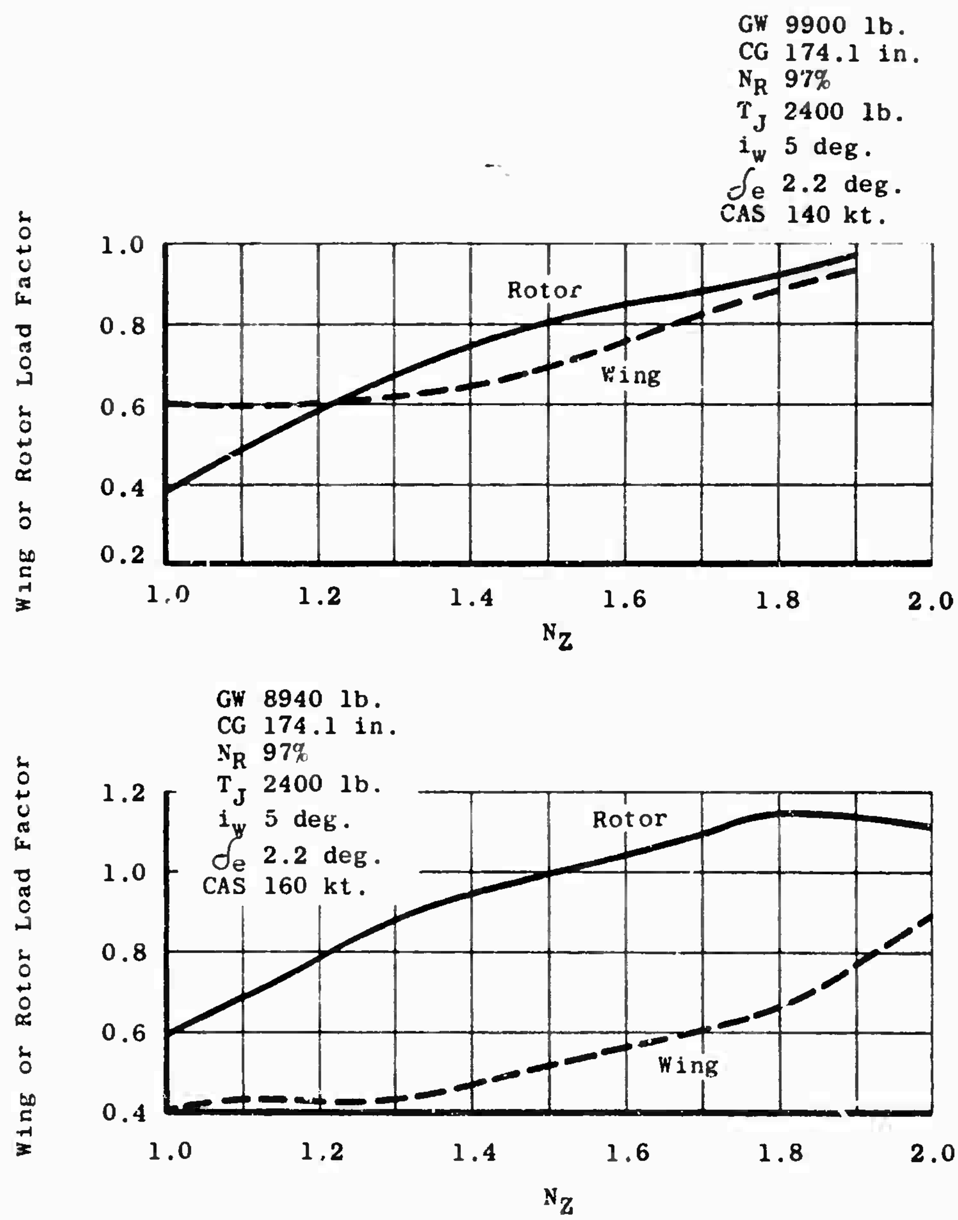

Figure 32. Wing and Rotor Load Factor Variation in Symmetrical Pull-Up Maneuvers.

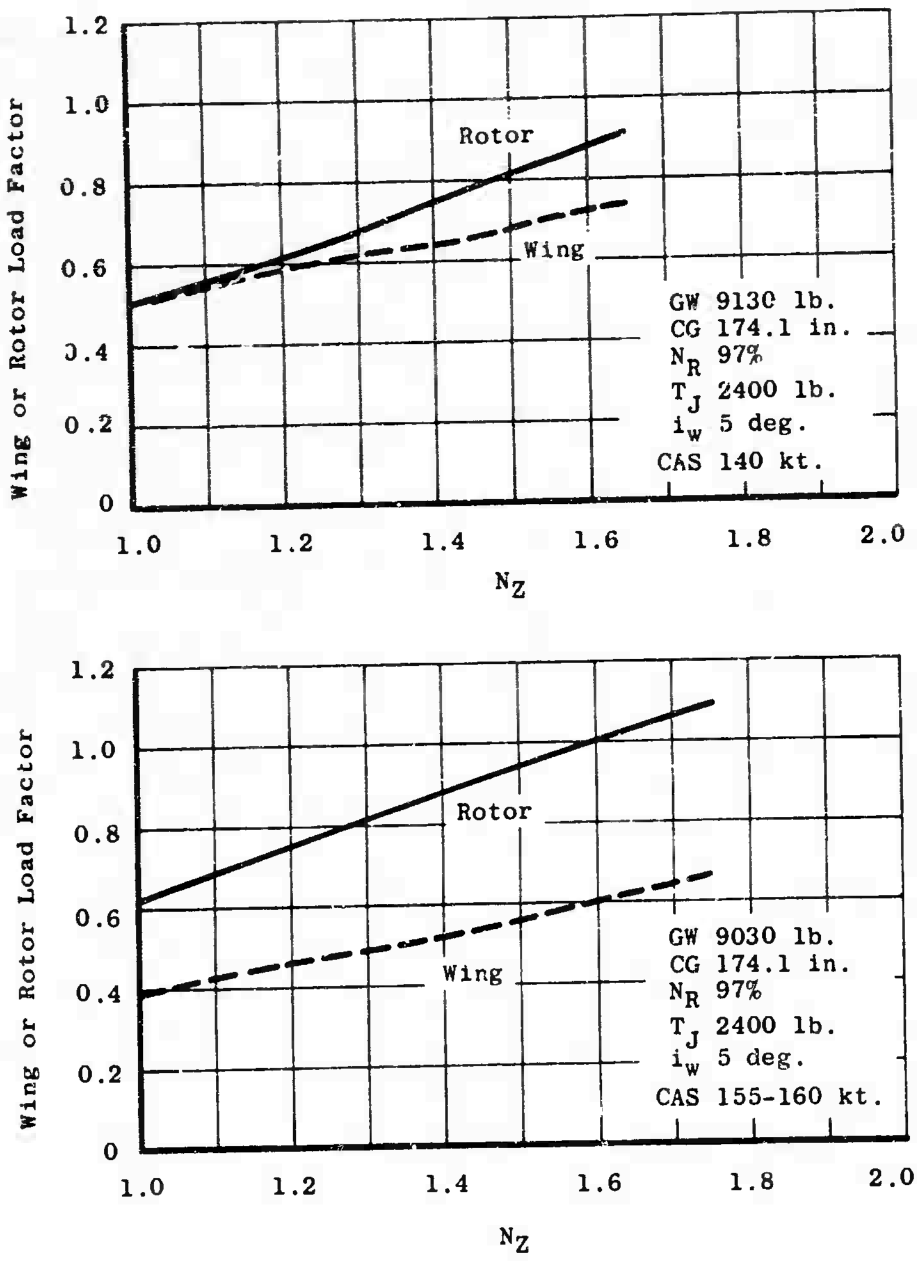

Figure 33. Wing and Rotor Load Factor Variation in Coordinated Right-Hand Turns, $\delta_e = 2.2$ deg.

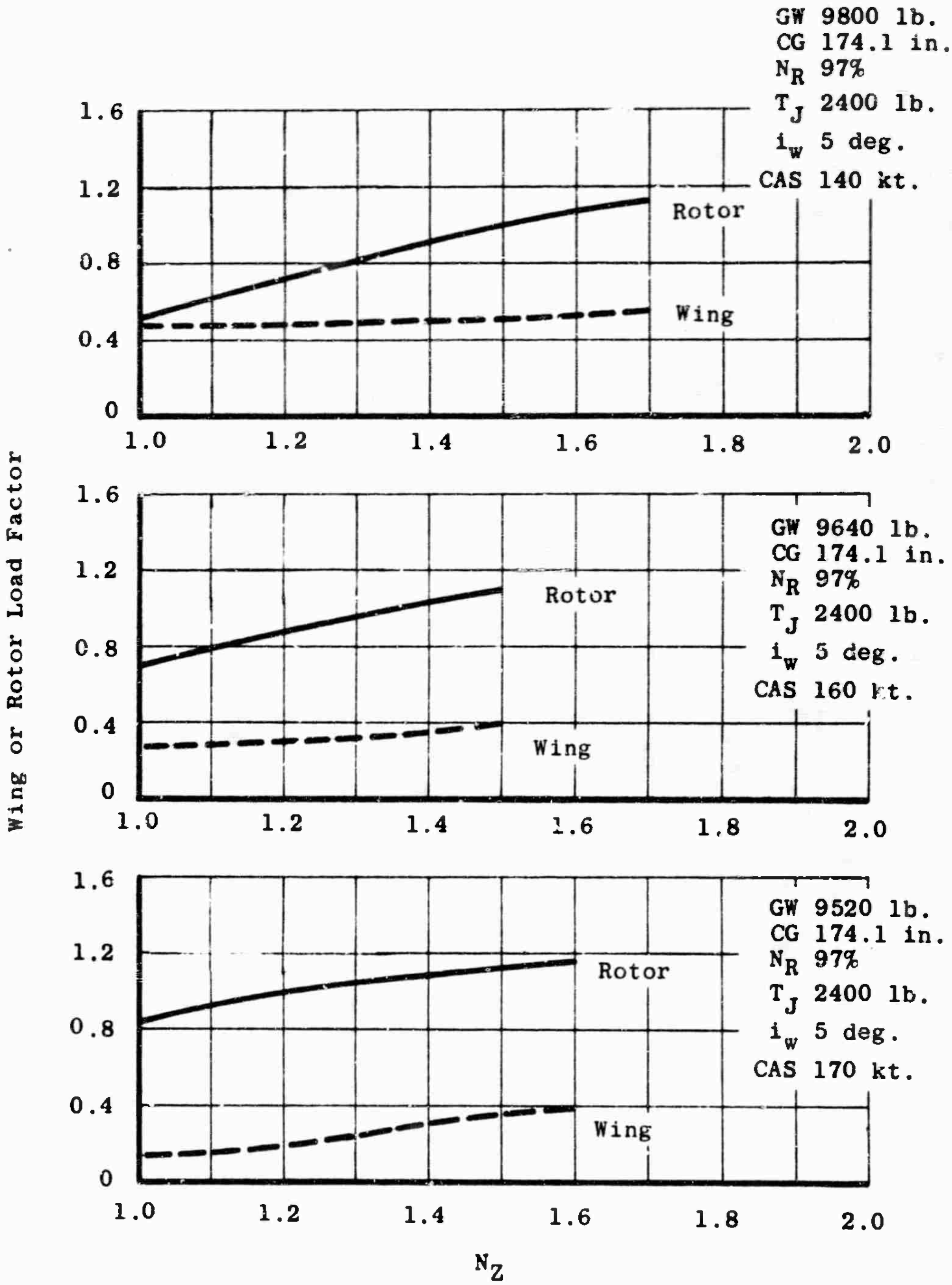

Figure 34. Wing and Rotor Load Factor Variation in Coordinated Right-Hand Turns, δ_e = 6.8 deg.

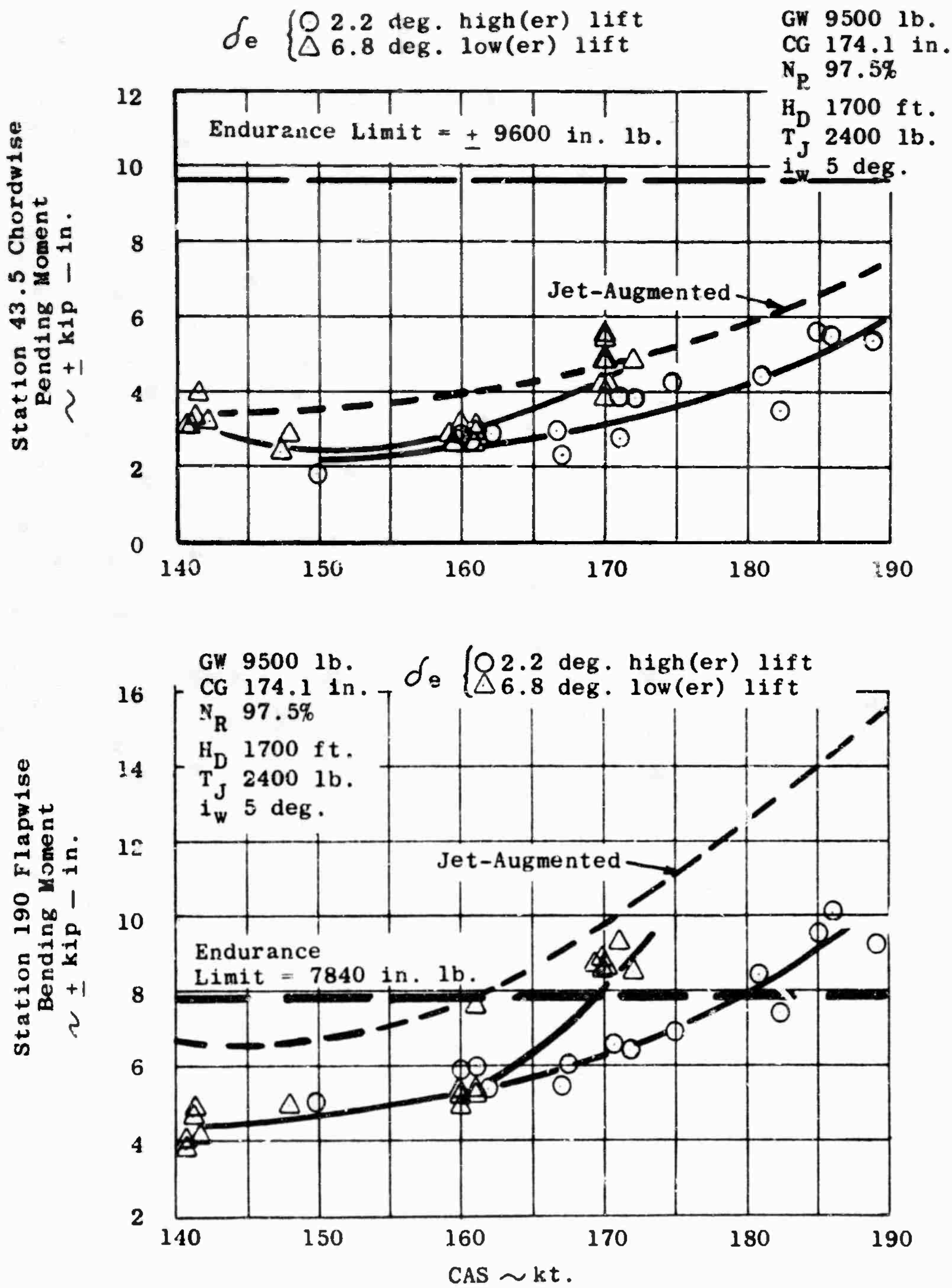

Figure 35. Effect of Wing Lift on Main Rotor Blade Bending.

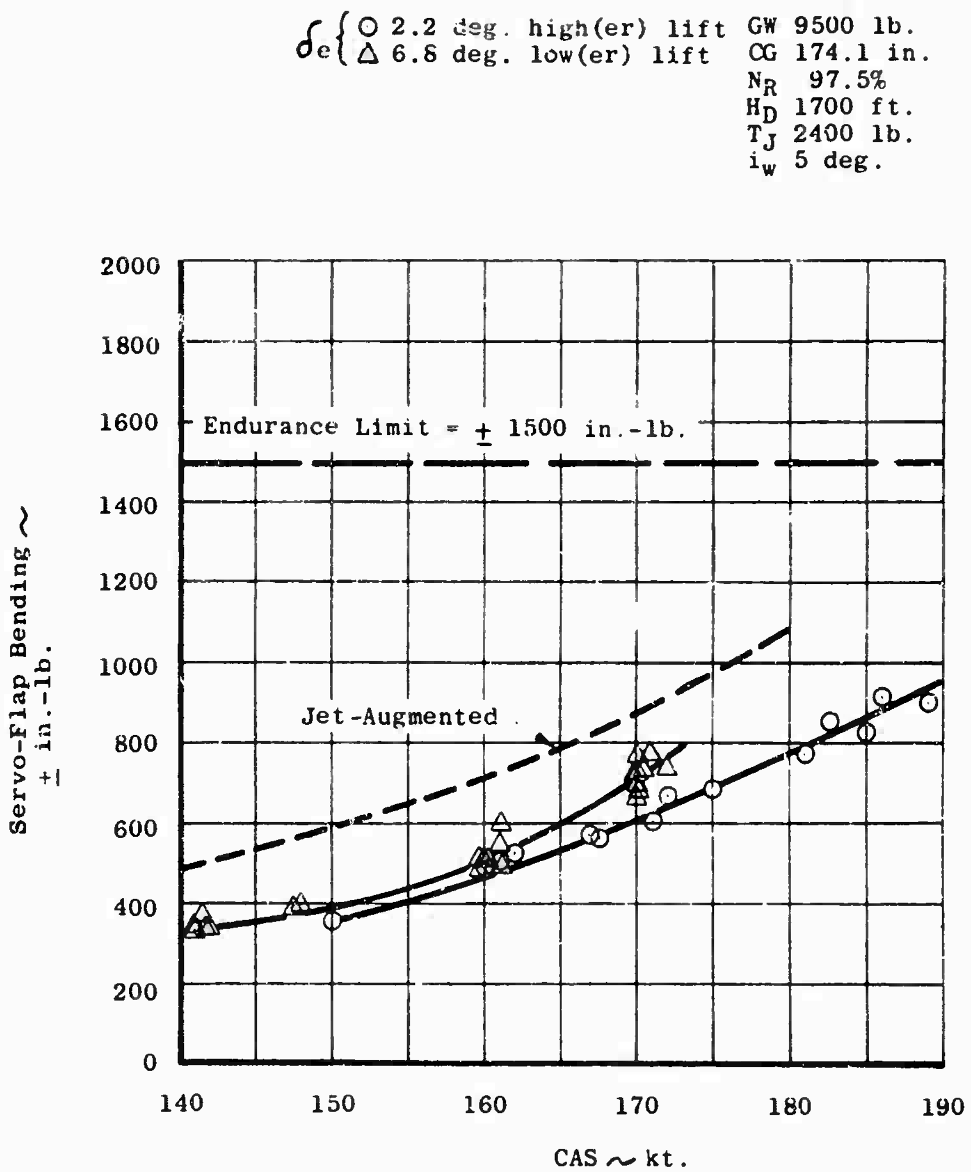

Figure 36. Effect of Wing Lift on Servo-Flap Bending.

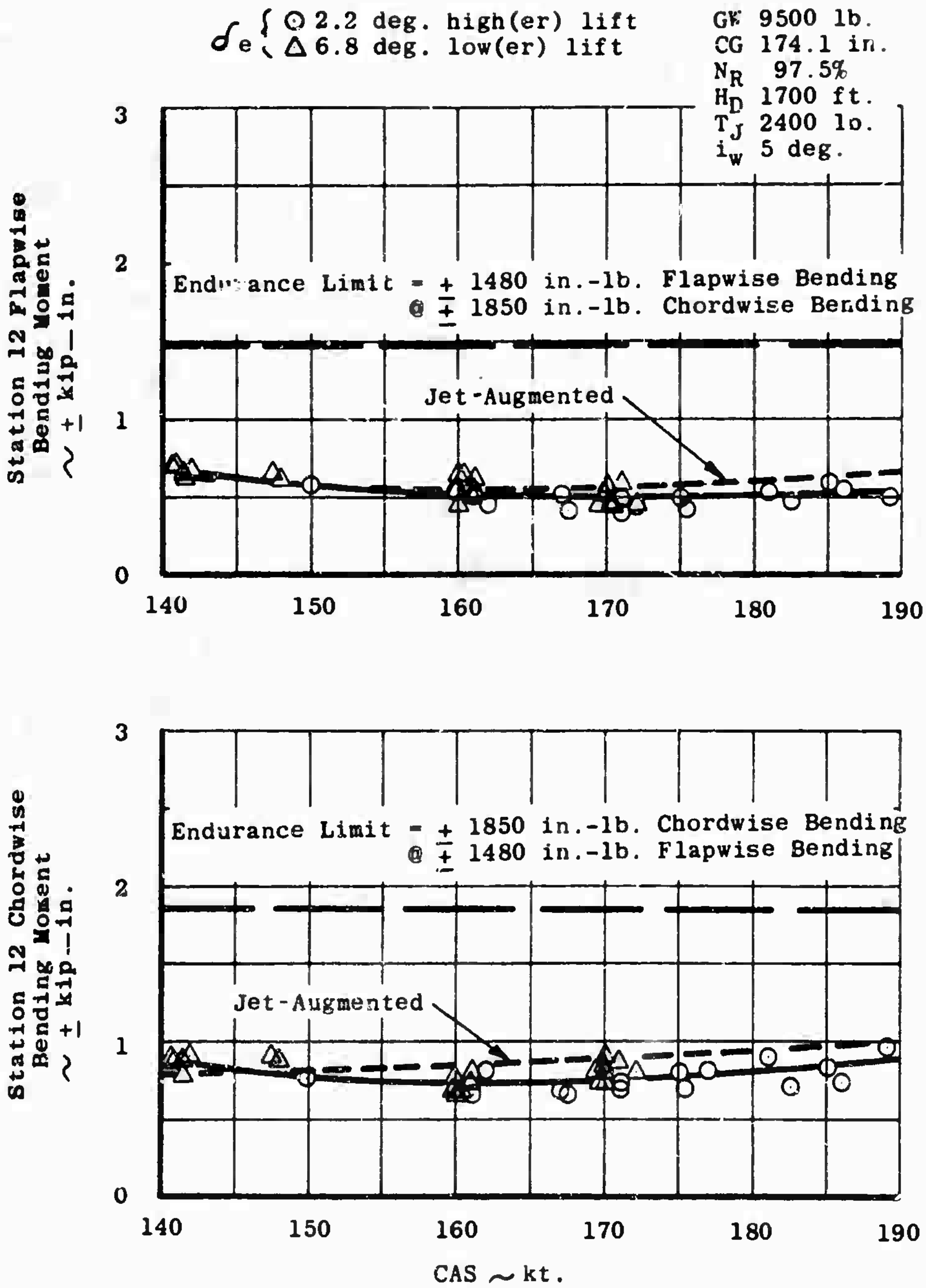

Figure 37. Effect of Wing Lift on Tail Rotor Blade Bending.

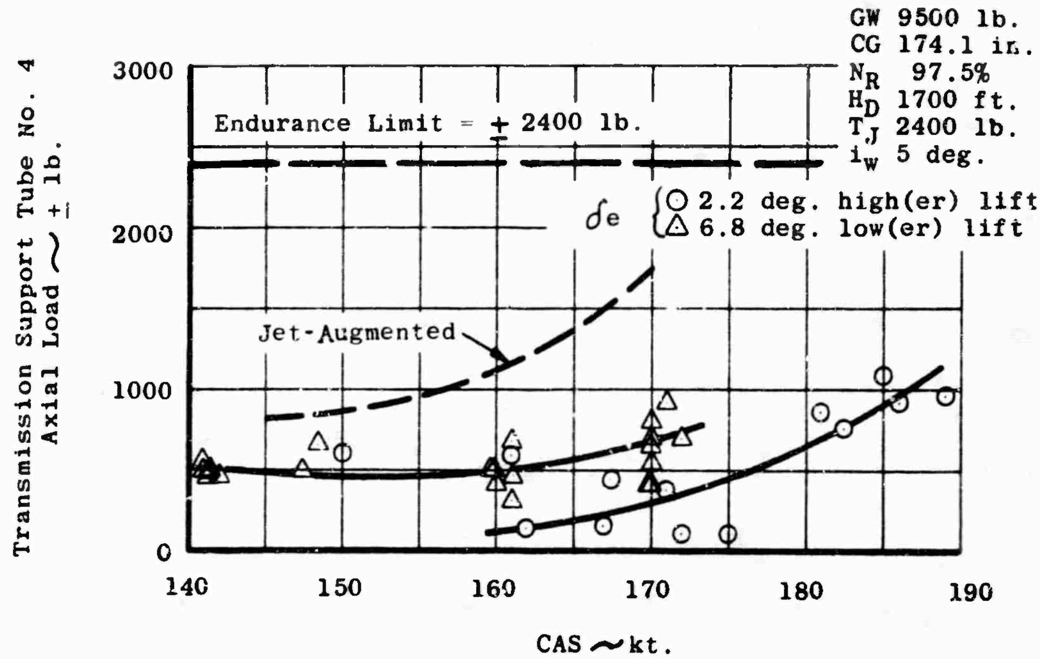

Figure 38. Transmission Support Tube No. 4 Vibratory Loads.

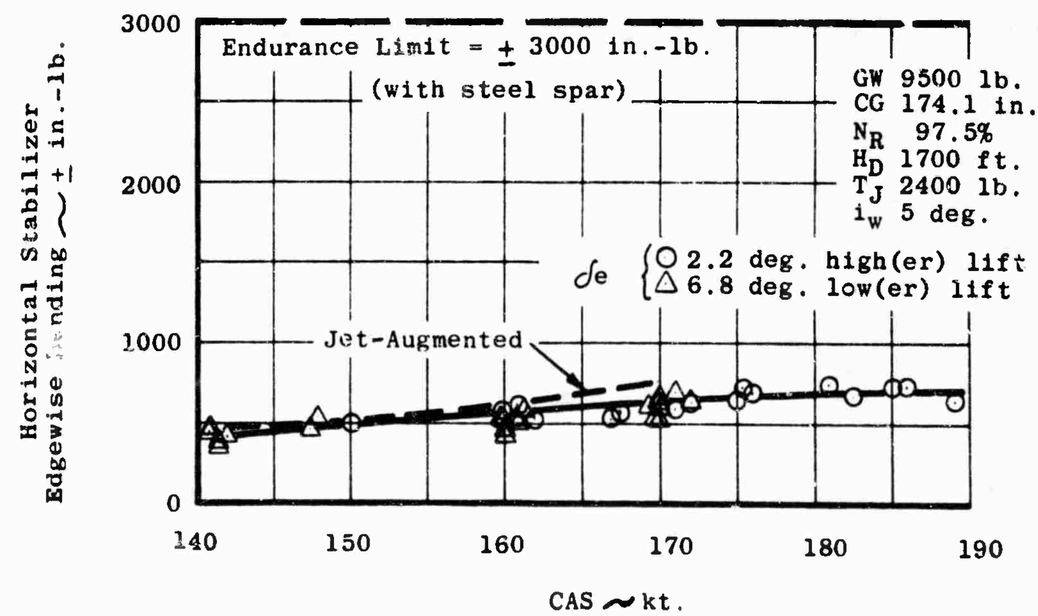

Figure 39. Horizontal Stabilizer Vibratory Bending.

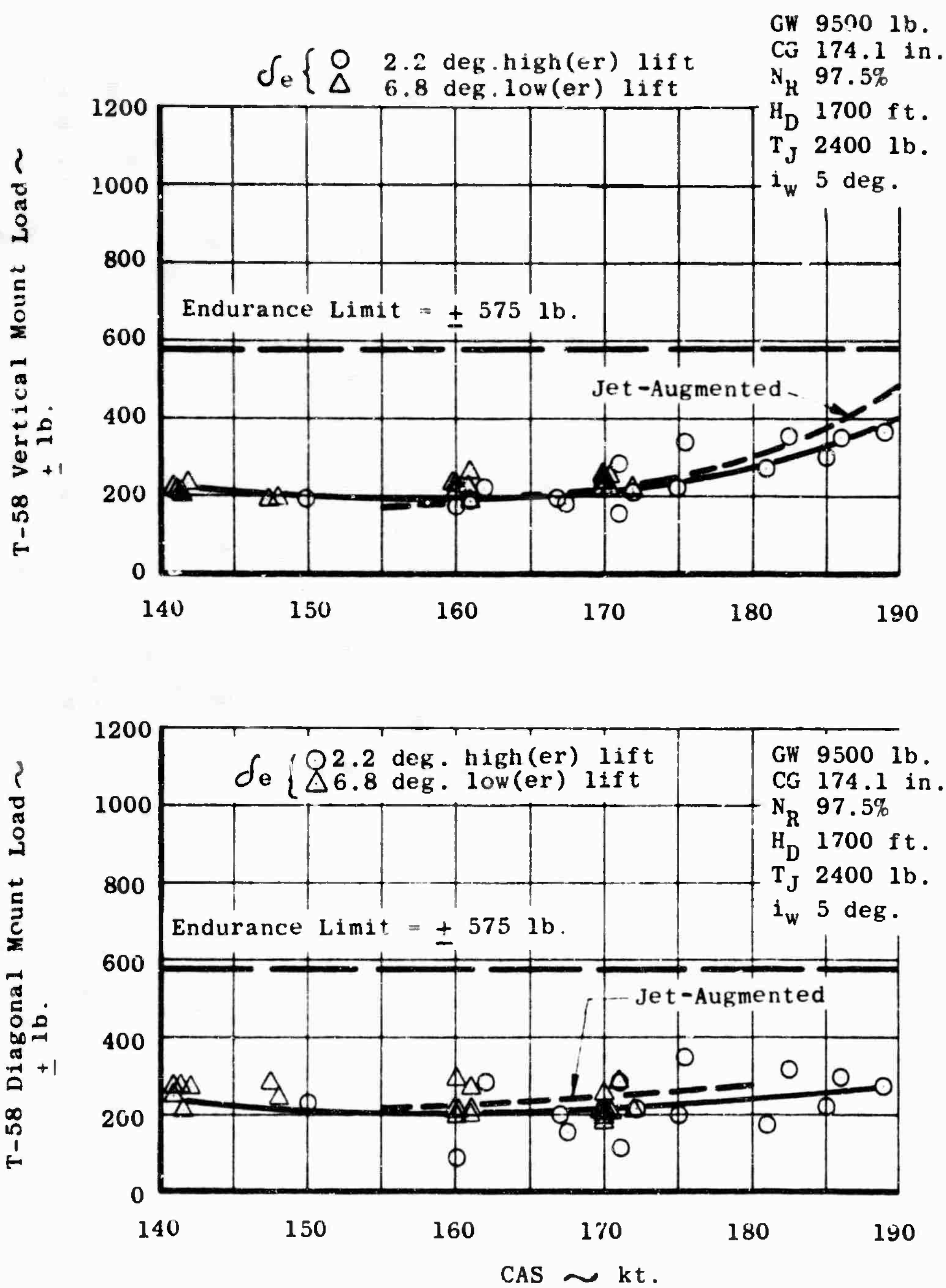

Figure 40. Effect of Wing Lift on T-58 Mount Vibratory Loads.

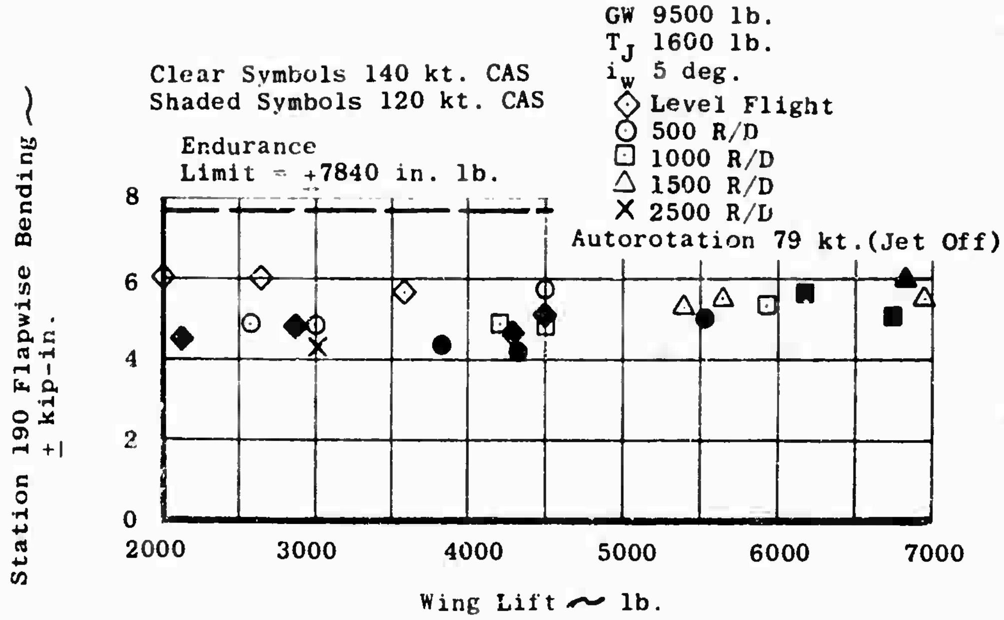

Figure 41. Main Rotor Flapwise Bending Versus Wing Lift and Rate of Descent.

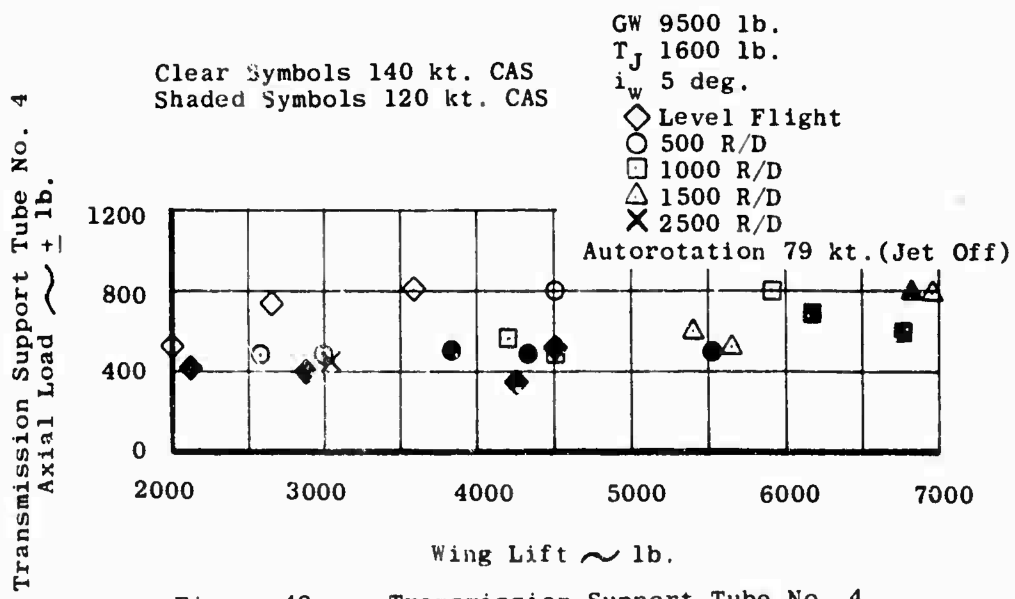

Figure 42. Transmission Support Tube No. 4 Vibratory Loading Versus Wing Lift and Rate of Descent.

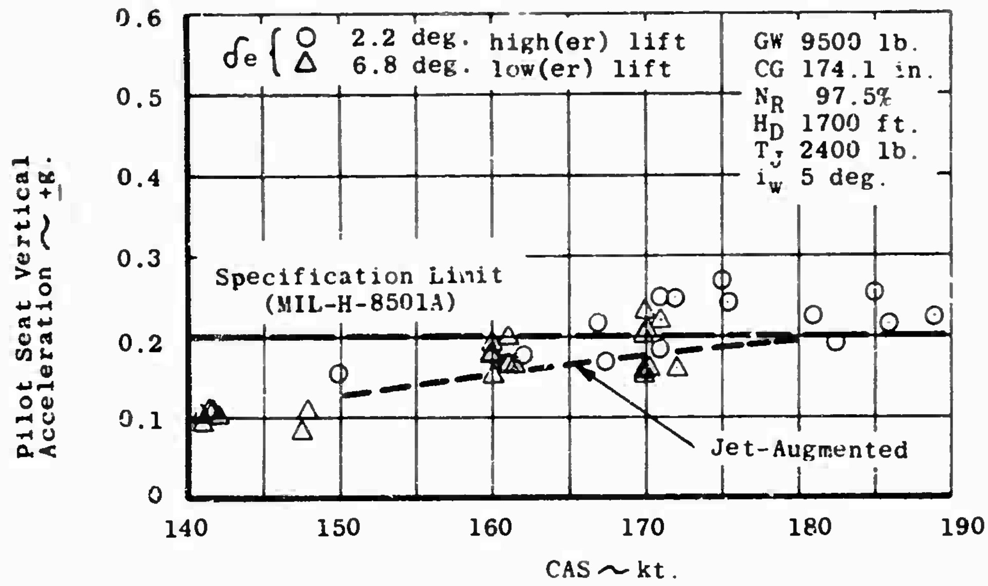

Figure 43. Pilot Seat Vertical Vibratory Acceleration.

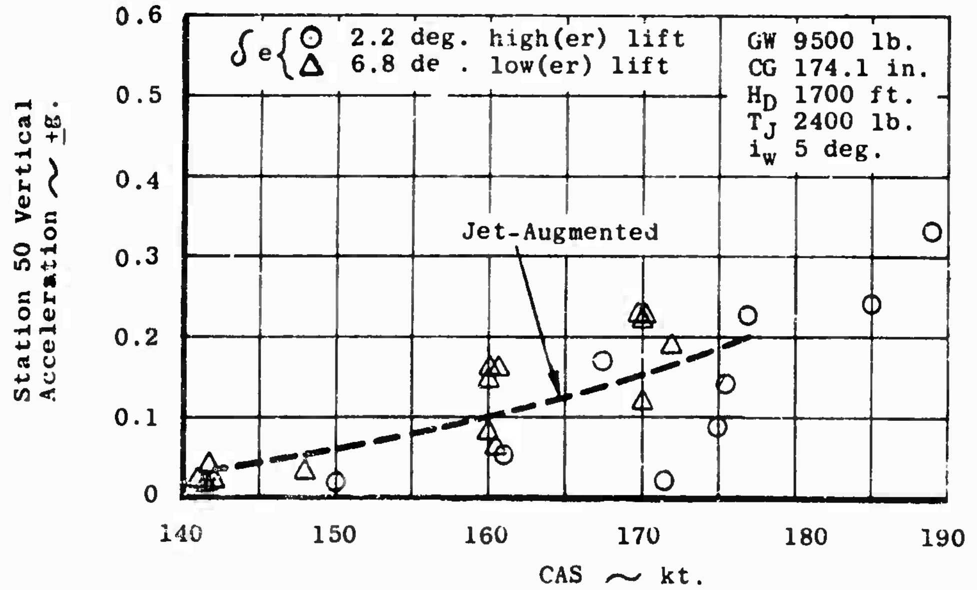

Figure 44. Station 50 Vertical Vibratory Acceleration.

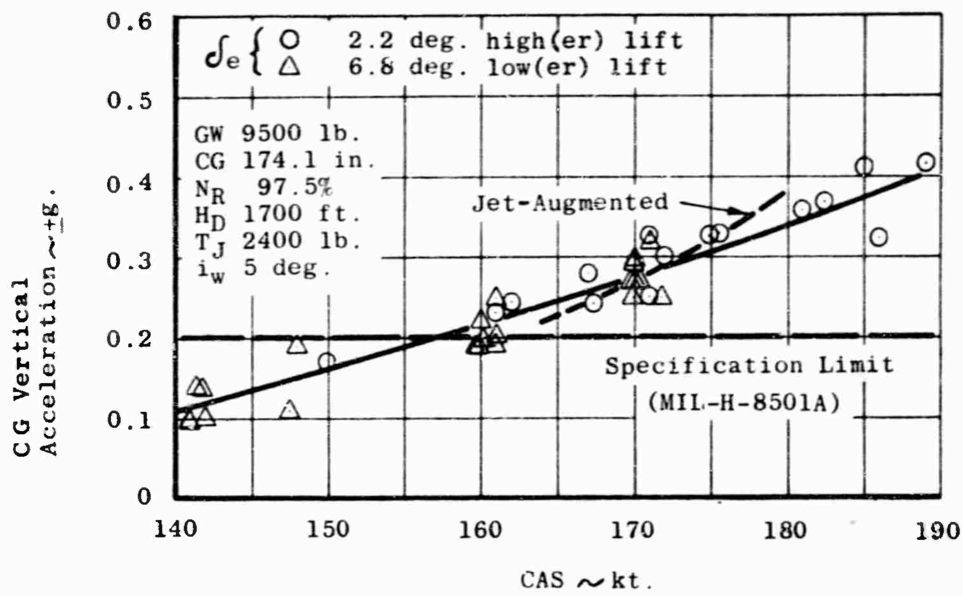

Figure 45. CG Vertical Vibratory Acceleration.

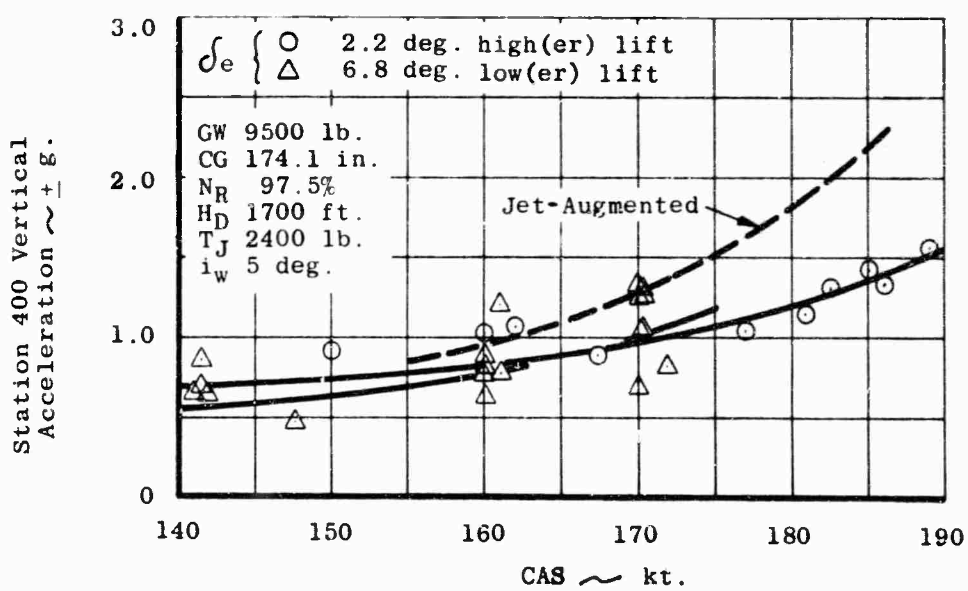

Figure 46. Station 400 Vertical Vibratory Acceleration.

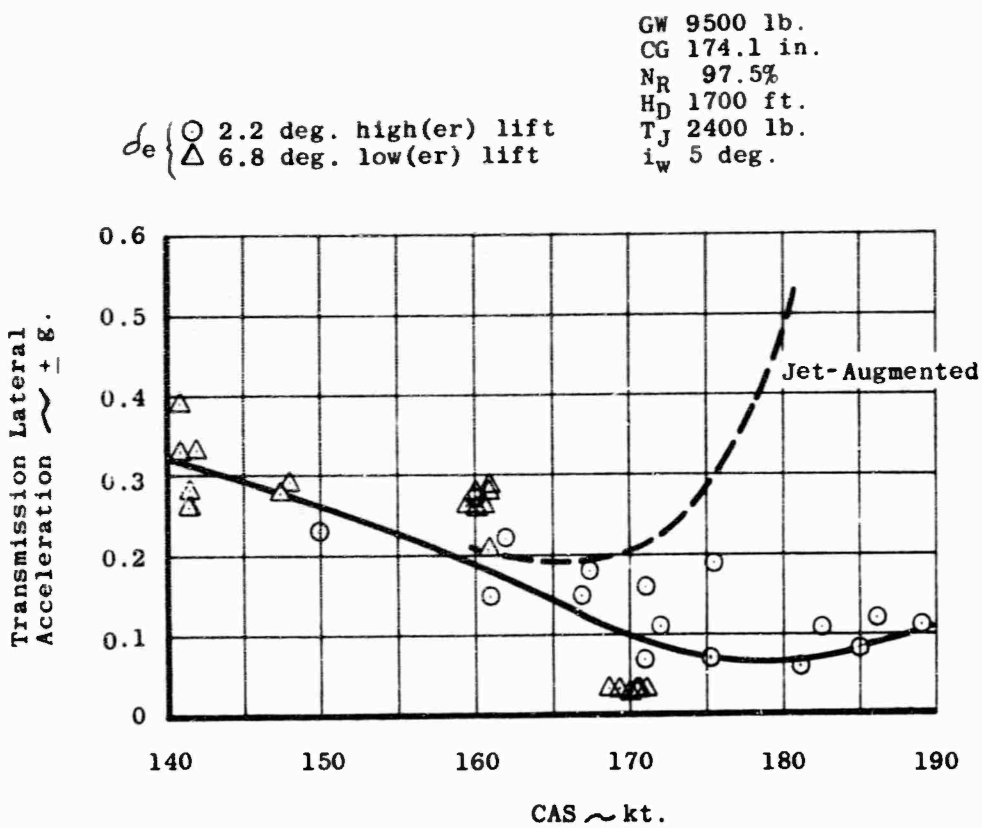

Figure 47. Transmission Lateral Vibratory Acceleration.

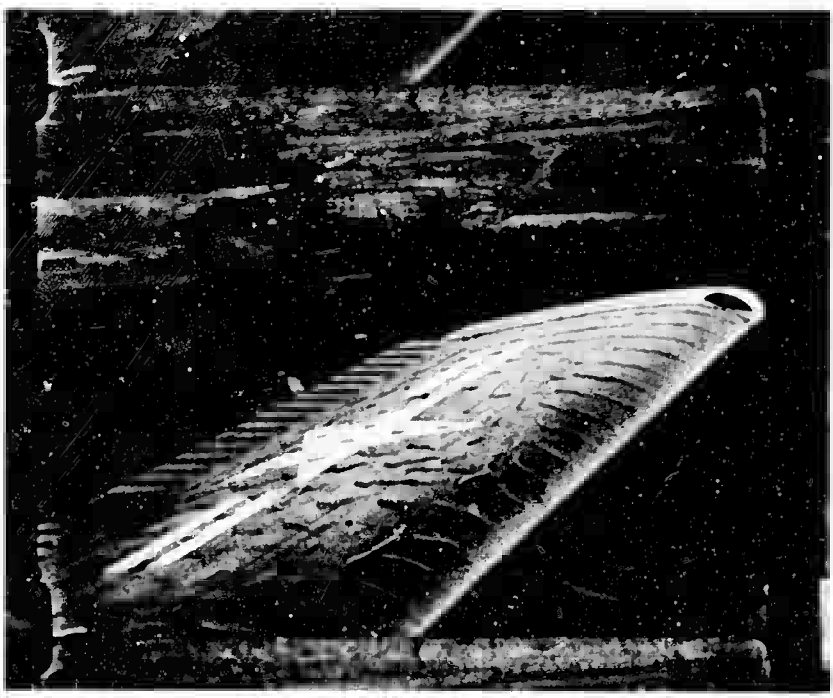

Figure 48.   Wing Tuft Behavior.

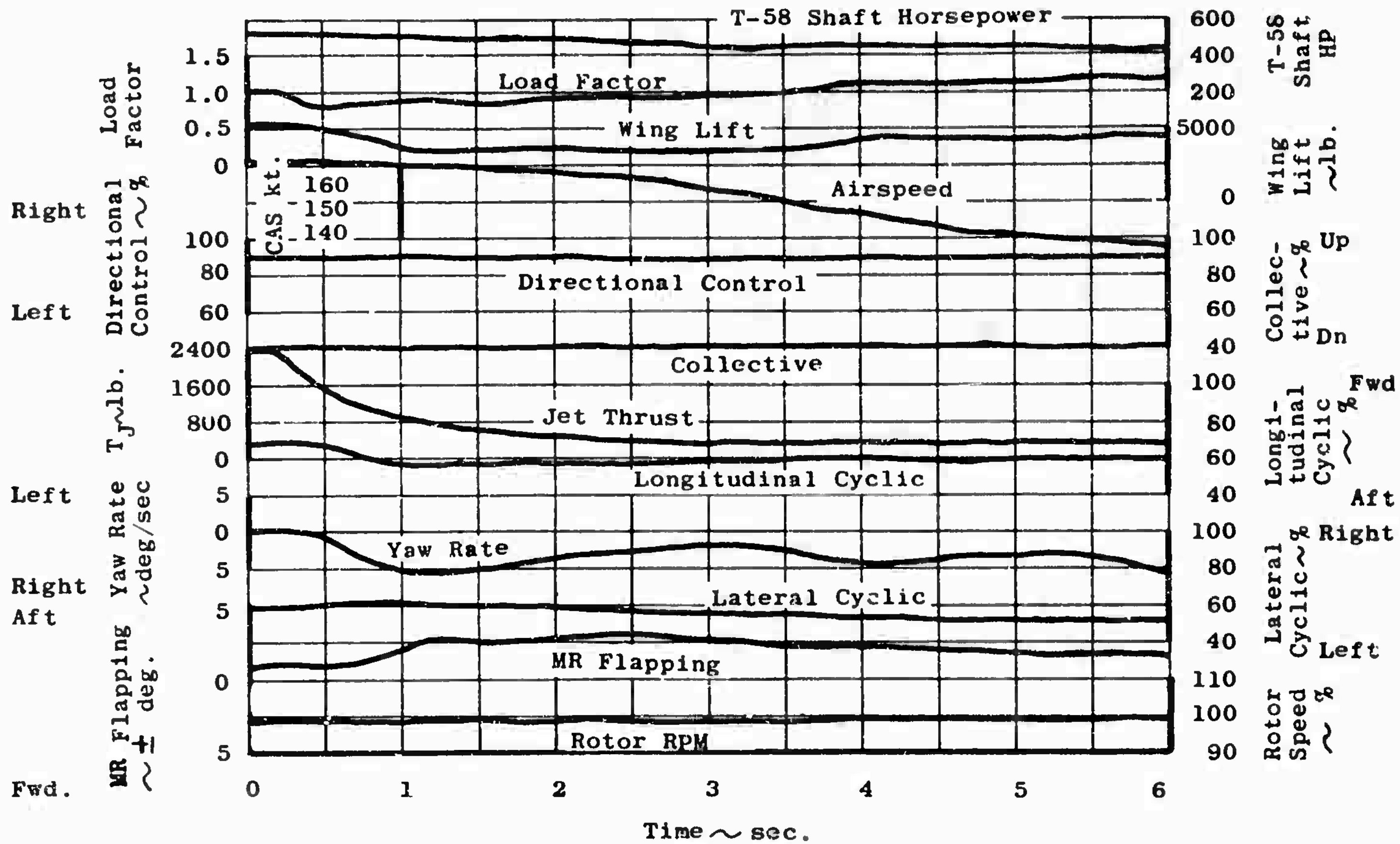

Figure 49. YJ-85 Throttle Chop From High Jet Thrust at 166 Knots TAS.

CONCLUSIONS

The use of auxiliary lifting surfaces is a valid concept for extending the speed and maneuver capability of rotary-wing aircraft. Lift augmentation substantially reduces main rotor and airframe loads and vibratory accelerations, and delays the onset of blade stall and compressibility.

The level-flight speed of the present configuration is limited by power available. A significant extension of the speed envelope can best be obtained by the addition of a second jet engine.

A wing with fixed incidence angle with the inherent advantages of rigid construction, reduced complexity, and maximum utilization can be incorporated in the design of compound aircraft. Wing area appears to be a compromise between requirements for maneuver capability at high speed and optimization of flying qualities in autorotation and low-speed maneuvers as well as aircraft performance.

Pure rotor control may not be optimum for compound helicopter maneuver at high speed. A means of reducing the portion of the load factor carried by the rotor in accelerated maneuvers is indicated, as well as augmentation of the roll control power supplied by the rotor. An integrated system of conventional airplane and rotor controls may be desirable.

Directional stability characteristics are essentially unaffected by lift augmentation. Lateral/directional response can be strongly influenced by the dihedral angle resulting from the combined effect of the wing and rotor. Longitudinal stability appears to approach the characteristics of fixed-wing configurations. Coupling of roll, pitch, and yaw motions can lead to apparent pitch instability in control-fixed maneuvers.

The response of the aircraft to sudden loss of jet thrust is easily controlled. The response of the aircraft to a loss of rotor power depends upon the amount of power being used, with recovery becoming increasingly difficult as rotor power increases.

Present analytical methods are, in general, satisfactory for predicting compound helicopter performance, trim and controllability, and limit airspeeds for the fully articulated servo-flap controlled rotor.

RECOMMENDATIONS

Based upon the results of this research program, it is recommended that:

Additional flight testing of the compound helicopter be conducted to investigate roll control with ailerons supplementing rotor control and to evaluate the use of a collective bobweight to favorably adjust rotor/wing load sharing during maneuvers.

An analytical and test program be conducted to study the influence of wing parameter changes on aircraft flight characteristics. The effect of wing area on autorotation, low speed handling qualities, rotor unloading, and aircraft stability should be included together with an examination of the influence of wing geometric dihedral angle on Dutch roll and pitch-yaw coupling.

It is also recommended that (from Reference 1):

Action be initiated to provide for the addition of greater thrust augmentation to develop the full potential of this research vehicle. This should make possible the acquisition of data in the 220- to 240-knot region.

An analytical and test program be conducted to evaluate potential blade tip section changes to increase the allowable tip Mach number prior to the onset of compressibility.

An analytical study be conducted in the area of control and power management for future rotary-wing aircraft, incorporating both horizontal thrust and lift augmentation. The objective of such a study would be to establish criteria for designing the system to integrate the pilot's power management and flight control activities. The study must take into account the flight requirements for future rotary-wing vehicles, as established by current research programs, and human factors considerations.

An analytical program be conducted to examine the effects of individual and simultaneous failure of the main and auxiliary power plants. Such a study would supplement the analysis conducted and the test results obtained to date. This study should include a determination of the characteristics of automatic devices which might be required to achieve satisfactory recovery from such a failure.

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13. ABSTRACT

The report presents results of a flight test program conducted using a UH-2 helicopter with a fully articulated, servo-flap controlled rotor and provided with a YJ-85 jet engine for thrust augmentation and with wings for lift augmentation. The effect of lift augmentation on airspeed limitations imposed by rotor blade stall or compressibility was investigated, and performance, flying qualities, structural loads, and vibrations were examined.

A maximum level-flight speed of 190 knots was achieved. It was determined that the airspeed is limited by power available. Although compressibility effects were encountered at high rotor speed, the airspeed envelope was expanded by reducing rotor speed.

Lift augmentation is shown to substantially reduce rotor and airframe loads and vibrations while providing an expanded speed and maneuvering envelope. The suitability of a fixed-wing incidence angle was demonstrated, although it is concluded that further investigation of rotor and aerodynamic surface controls may be required to provide optimum control.

Correlation of flight test results and those predicted by analytical study is presented in areas of performance, controllability, and limit flight speeds. In general, analytical methods for predicting these characteristics are shown to be satisfactory.

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14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
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